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SPACE TRANSPORTATION BOOSTER ENGINE CONFIGURATION STUDY

FINAL REPORT (DR4)

INCLUDES

DESIGN DEFINITION DOCUMENT (DR8)

AND

ENVIRONMENTAL ANALYSIS (DR10)

31 MARCH 1989

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FOREWORD

This study was conducted by the Pratt & Whitney/Government Engine Business (P&W/GEB) of the United Technologies Corporation under NASA/MSFC contract NAS8-36857. The NASA/MSFC program manager was Mr. J. Thomson. The Pratt & Whitney program manager was Mr. W. A. Visek, Jr., and D. R. Connell was the booster engine program manager.

The technical effort started in March 1986 and was completed in March 1989. The study is presented in three volumes.

Volume I — Executive Summary
Volume II — Final Report
Volume III — Program Cost Estimates

Special thanks go to the numerous individuals at NASA, UTC, and the major vehicle contractors who contributed to this study effort.

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INTRODUCTION

The United States is experiencing a critical need to place large payloads in low earth orbit. This need exceeds the capability of current and planned fleets of Titan IV and Space Shuttle launch vehicles, and reflects the requirements of the National Aeronautics and Space Administration (NASA), the U. S. Air Force, the Strategic Defense Initiative Organization (SDIO), and the civilian sector.

The Advanced Launch System (ALS) will provide a low cost, reliable means of satisfying this need. The ALS will enable the United States to meet defense, national, and civil launch requirements, while expending fewer resources on launch vehicles.

The objective of the Space Transportation Booster Engine Configuration Study is to contribute to the ALS development effort by providing highly reliable, low cost booster engine concepts for both expendable and reusable rocket engines.

An artist's concept of a fully reusable booster with a partially reusable core vehicle is shown in Figure 1.

The objectives of the Space Transportation Booster Engine (STBE) Configuration Study were: (1) to identify engine configurations which enhance vehicle performance and provide operational flexibility at low cost, and (2) to explore innovative approaches to the follow-on Full-Scale Development (FSD) phase for the STBE.

The Pratt & Whitney (P&W) overall technical approach to the study, shown in Figure 2, was based on the STBE technical requirements and guidelines presented in the Statement of Work (SOW). These requirements and guidelines were modified continually as the results of the joint NASA/Air Force Space Transportation Architecture Study (STAS), and later the Advanced Launch System (ALS), became available. As a result, the study effort was completely supportive of and interactive with the ALS and other launch vehicle studies. The schedule of the STBE Phase A, including the three extensions and the interim final reporting documentation, is shown in Figure 3.

The STBE Configuration Study consisted of six tasks. Task I (SOW Task 5.1) consisted of parametric analyses and trade studies. First, the system design requirements and features were defined, and the information base was established. Second, the STBE configurations that enhance performance and provide operational flexibility at low cost were identified, and the requirements for those engine configurations for the projected missions were defined.

During Task II (SOW Task 5.2), P&W developed a plan to evaluate the STBE configurations identified in Task I and established criteria to select the most promising configurations. The Configuration Evaluation and Criteria Plan used overall system life cycle costs as the figure of merit and included considerations of mission and vehicle requirements, operational flexibility, schedules (along with their risks), required technological advances, and facility requirements. The evaluation and selection criteria were compatible with the NASA requirements and the STAS results.

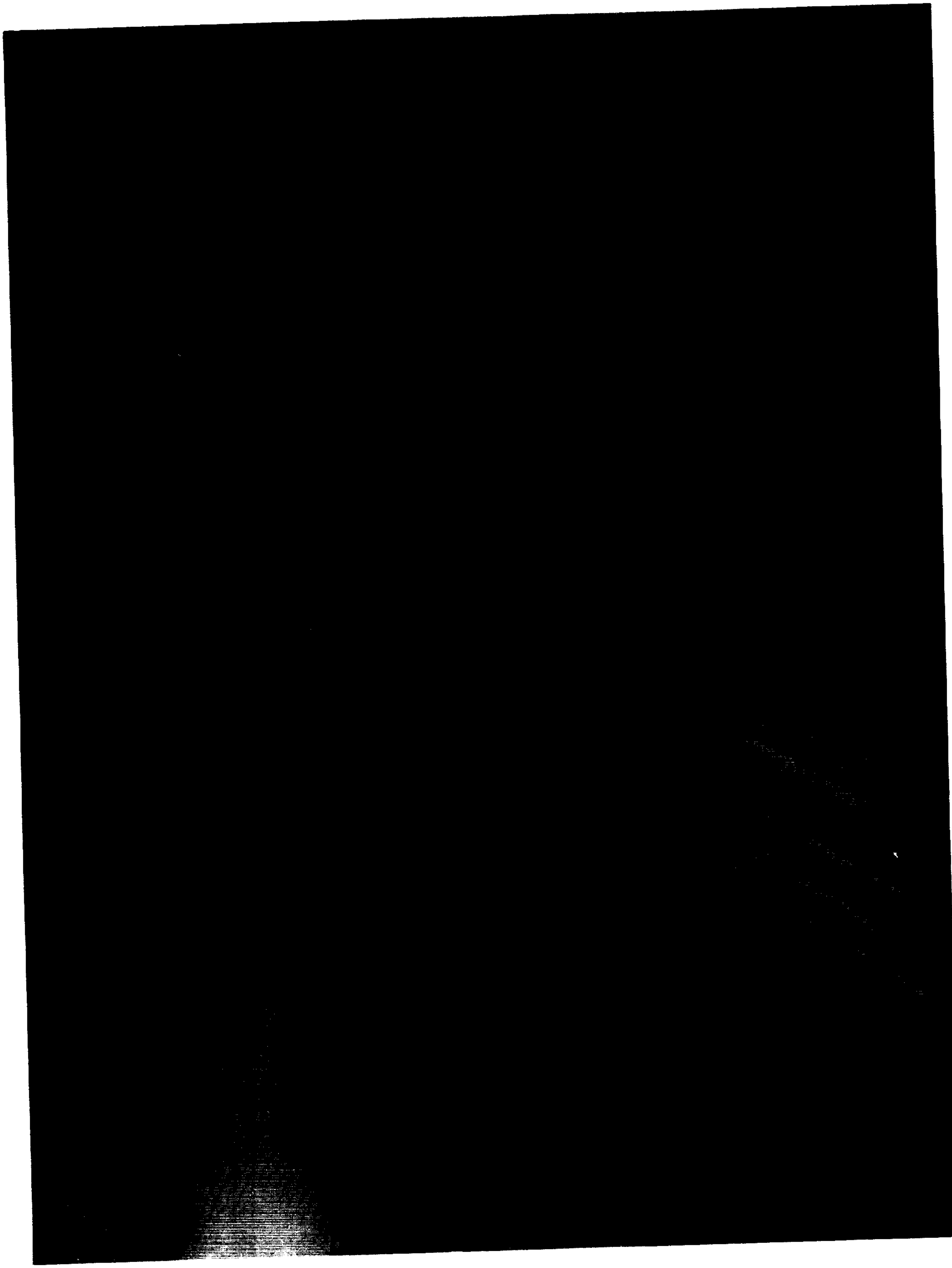

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Figure 1. Artist's Concept — Fully Reusable Booster With Partially Reusable Core Vehicle

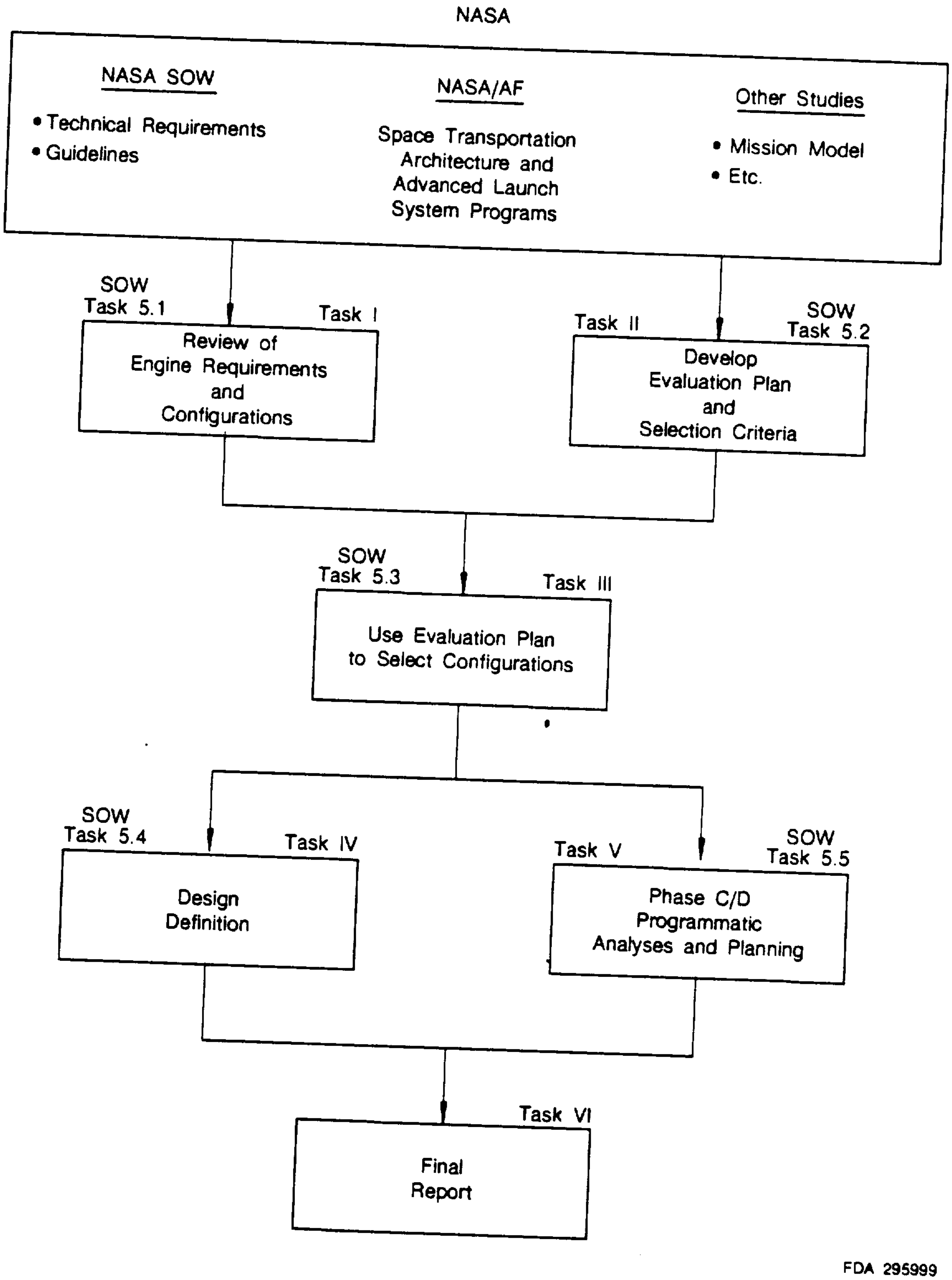

Figure 2. Overall Approach to Space Transportation Booster Engine Configuration Study

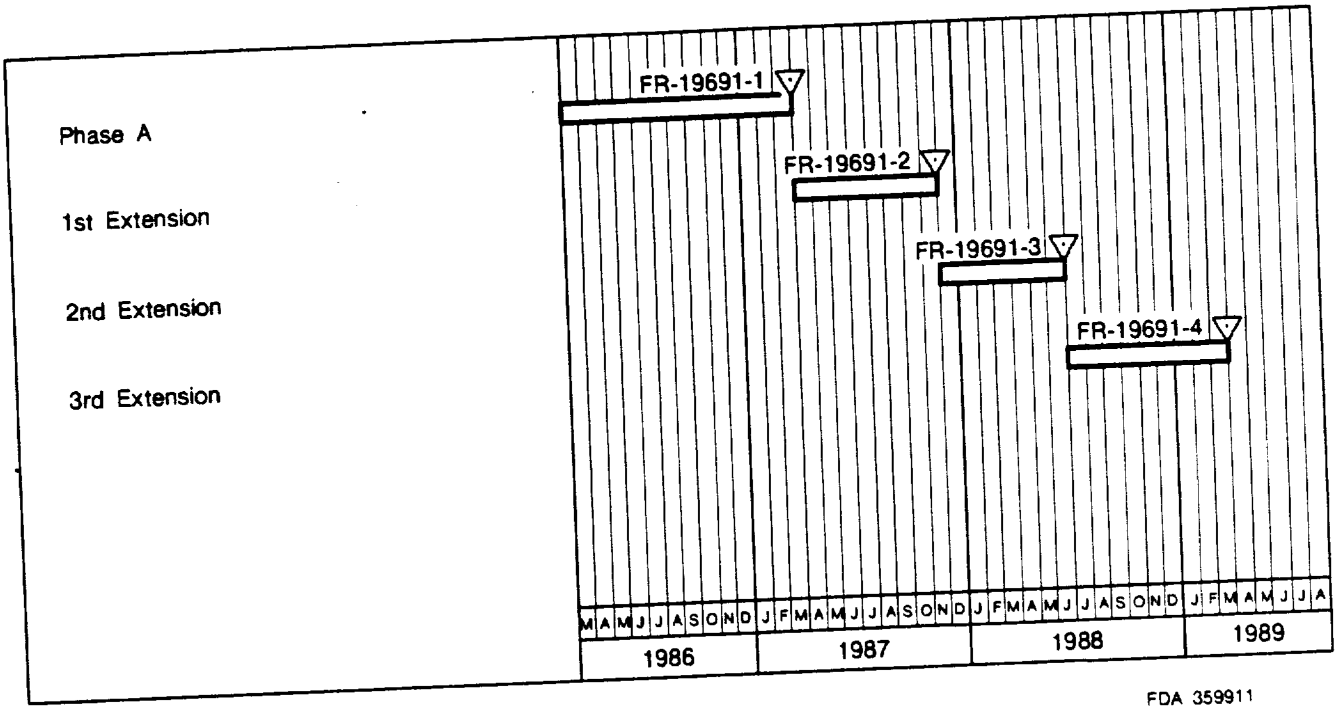

Figure 3. STBE Phase A and Extensions

During Task III (SOW Task 5.3), P&W assessed the STBE configurations and requirements identified during Task I using the Configuration Evaluation and Criteria Plan developed during Task II. This process, based on minimizing life cycle cost (LCC), was used to select the most promising engine candidate as agreed to by NASA and P&W.

The selected engine candidate was then the subject of Tasks IV and V. During Task IV (SOW Task 5.4), P&W completed the conceptual designs of the selected candidate. Under this task, P&W prepared the Design Definition Document (DR8), including a preliminary Interface Control Document (ICD) and preliminary Contract End Item (CEI) Specification. Task V (SOW Task 5.5) was conducted concurrently with Task IV and provided the plans for FSD. These plans included schedules, facility requirements, a Work Breakdown Structure (WBS) and dictionary, a cost analysis, and an Environmental Impact Analysis (DR10).

During Task VI, all of the technical reviews, status reports, and the final report were prepared.

The Interim Preliminary Reports were published at the milestones shown in Figure 3. The information and studies reported within these documents are referenced but not repeated in this Final Report.

This Volume II, Final Report, contains all of the work conducted under the contract during the time period July 1, 1988 to March 31,1989. The first section (1.0 - Evolution of STBE Phase A) provides a narrative of the STBE Phase A effort that ties together the reference documents.

All costs contained in this volume are engineering estimates. These costs should not be considered as contractual commitments and should be used for Life Cycle Cost (LCC) evaluations and planning purposes only.

The STBE Program WBS and cost estimates are presented in Volume III.

SECTION 1.0
EVOLUTION OF STBE DURING PHASE A

The Space Transportation Booster Engine (STBE) configuration study evolved over the three-year contract period. A brief overview of the significant phases of the study is shown in Figure 1-1.

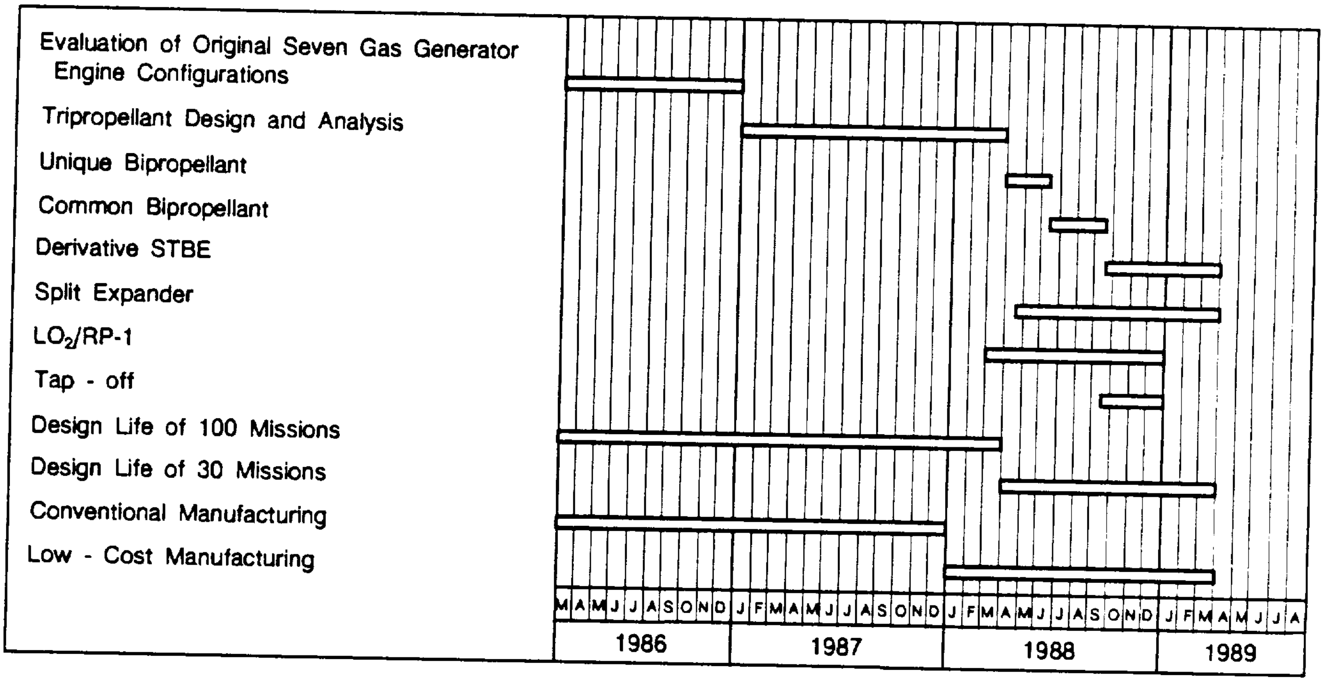

Figure 1-1. STBE Study Significant Phases

Seven Gas Generator engine configurations were initially identified that met the requirements set forth in Task 1, Vol. II of FR-19691-1. Their characteristics are given in Table 1-1. These configurations were assessed using the Configuration Evaluation and Criteria Plan developed during Task II. The engine evaluation process was based on determining the total life cycle cost (LCC) of a launch system using the ground rules for the trajectory, the vehicle, and for the programmatic considerations. In recent years, LCC has become the accepted standard criteria on which to make the "best" choice because it includes all the important elements of engine evaluation criteria: performance, weight, development difficulty, risk, and operations as well as cost. LCC is the figure of merit which encompasses the total system, and therefore requires system level analysis.

Figure 1-2 shows an overview of the launch vehicle/rocket engine optimization procedure that was used as the basis for the present study. After the study ground rules were established, the matrix of design variable (parameter) combinations was selected. Engine performance and weight were then calculated for each of the variable combinations. The vehicle characteristics were obtained by an iterative procedure that loops through the Vehicle Weight and Sizing Program, the Trajectory Program interface, and the engine performance and weight data, until a converged mission-capable vehicle was defined. The characteristics of this vehicle were then passed on to the Vehicle LCC Program, which also receives input from the Engine LCC Program. For each vehicle in the parametric matrix, LCC and weight data are passed into the Regression

Program which fits a multivariable surface defining LCC as a function of the design variables. The Optimizer Program then interrogates the surface and searches for the combination of design variables which results in a minimum LCC vehicle.

Table 1-1. *STBE Candidate Engine Configurations — All Gas Generator Cycles*

	STBE-1A	STBE-1B	STBE-2	STBE-3	STBE-4	STBE-5	STBE-6
Propellants	LO_2/RP-1	LO_2/RP-1	LO_2/RP-1	LO_2/CH_4	LO_2/CH_4	LO_2/C_3H_8	LO_2/C_3H_8
Coolant	RP-1	LO_2	LH_2	CH_4	LH_2	C_3H_8	LH_2
Mixture Ratio	2.90	2.90	3.12	3.57	3.64	3.20	3.38
Chamber Press (psia)	1275	1667	3500	2333	3500	2333	3500
Thrust							
Vacuum (lbf)	736,100	735,900	706,000	713,100	705,800	715,100	705,800
Sea Level (lbf)	625,000	625,000	625,000	625,000	625,000	625,000	625,000
Specific Impulse							
Vacuum (sec)	316.0	318.4	360.1	341.5	369.5	333.9	363.2
Sea Level (sec)	264.3	273.5	318.2	302.6	326.5	291.4	321.0
Area Ratio	25	35	55	40	55	40	55
Length (in.)	152	155	143	143	143	143	143
Diameter (in.)	98	98	84	88	84	88	84
Weight (lb)	6750	6745	6925	6655	6845	6650	6885

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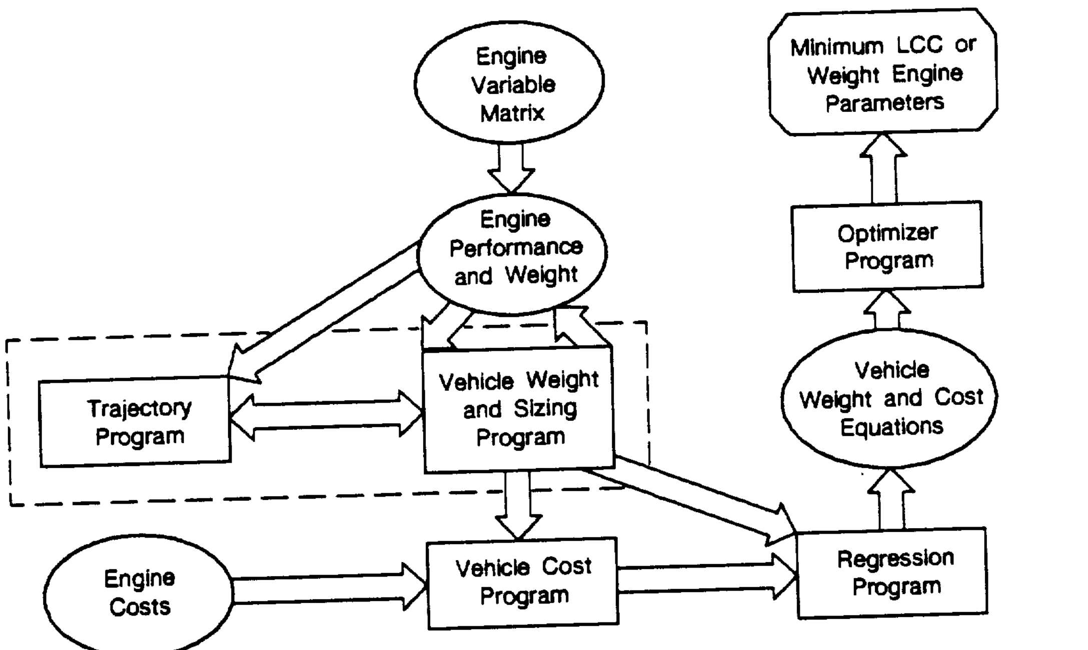

Figure 1-2. Launch Vehicle/Rocket Engine Optimization Procedure

The ground rules of this evaluation procedure were established jointly by Pratt & Whitney and the NASA Program Manager. Figure 1-3 describes the Launch Vehicle used in the analysis. A glideback booster with a 3000 ft/sec relative burnout velocity and a flyback booster with a 6000 ft/sec burnout velocity were evaluated to see if the optimum STBE characteristics changed. The trajectory ground rules are presented in Table 1-2, and the programmatic ground rules are presented in Table 1-3.

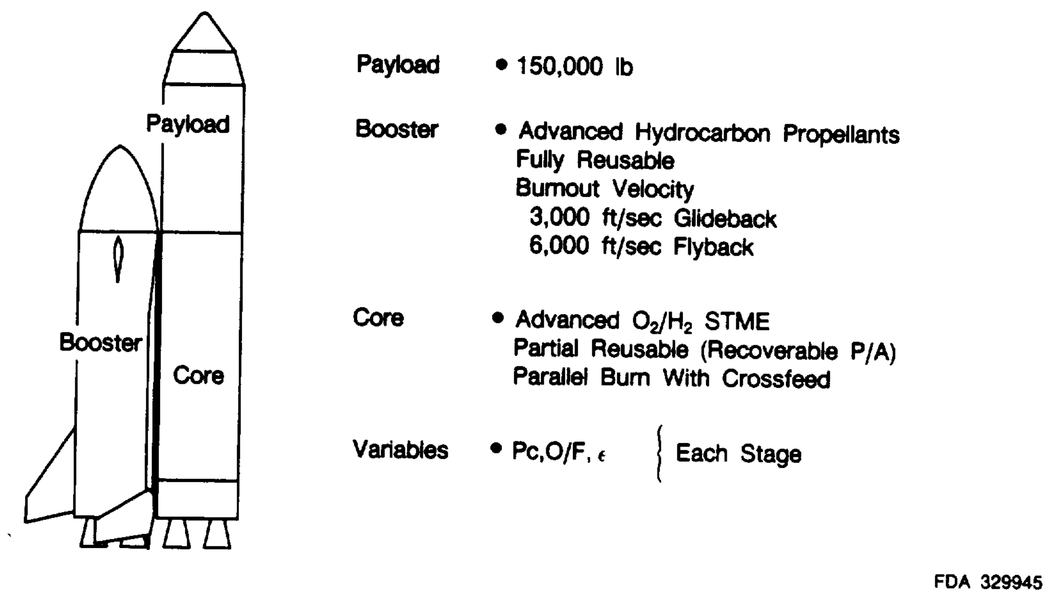

Figure 1-3. STBE Study — Launch Vehicle Used in Evaluation Analysis

Table 1-2. *STBE Study — Trajectory Ground Rules*

Trajectory Ground Rules Were as Follows:

- East Launch/28.5° Inclination
- Powered Ascent to 75 × 150 nm Orbit
- Circularize at Apogee Using OMS
- Maximum Q < 1100 lb/sq ft
- Maximum G < 4.0 (Axial)
- Optimized Pitch Schedule

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Table 1-3. *STBE Study — Programmatic Ground Rules*

	Flyback Booster	*Core Vehicle*
Active Number Vehicles	8	8
Avg Launch/Year/Vehicle	6	6
Development Time		
— Engine	7 yr	7 yr
— Vehicle	6 yr	6 yr
First Flight	1995	1995
Operating Period	15 yr	15 yr
Vehicle Useful Life	100 Missions	100 Missions

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The detailed discussion of this assessment and the results are presented in Volume II of FR-19691-1. The engine configuration selected by this process to have the lowest life cycle cost was the LO_2/methane/hydrogen Tripropellant Gas Generator.

The following factors that make the LO_2/methane/hydrogen tripropellant the lowest life cycle cost engine configuration also make good engineering sense:

- *Combustion Stability* — Methane has the best history of combustion stability of all of the hydrocarbon rocket fuels. Also, the addition of hydrogen into the main combustion process will increase the burning rate. This increase in burning rate is predicted to make the combustion process even more stable.

- *Combustion Efficiency* — Although high combustion efficiencies have been obtained in LO_2/methane combustion systems, adding hydrogen to the combustion process increases the calculated combustion efficiency for the various hydrocarbon fuels for a resultant higher efficiency than LO_2/methane.

- *Cooling Capability* — The excellent cooling capability of liquid hydrogen has been established in several operational engine designs.

- *Ignition* — An oxygen/hydrogen torch igniter can be used. The ignition limits of oxygen and hydrogen are very broad. This allows ignition at low pressures and mixture ratios well away from the stoichiometric mixture ratio. The hydrogen/oxygen ignition source also heats the methane/oxygen mixture for easier main chamber ignition. The main chamber could also be started with only oxygen and hydrogen.

- *Cleanliness of Turbine Drive Gas* — The exhaust of the gas generator driving the turbines is hydrogen and steam; it is clean, and is used successfully in the Space Shuttle Main Engine.

- *Chamber Material Compatibility* — Hydrogen is known to be compatible with the copper alloys used in the design of combustion chambers. However, because of hydrogen embrittlement the usual care must be taken in the selection of materials.

- *Safety* — Both methane and hydrogen are lighter than air at ambient pressure and temperature, therefore, leaks or spills will not accumulate in low areas.

In summary, the selection of the LO_2/methane fueled, hydrogen cooled tripropellant engine configuration either eliminates or greatly reduces the risks associated with the design of high pressure, reusable hydrocarbon booster engines.

This tripropellant engine configuration selection was then carried into Tasks IV and V. During Task IV the conceptual design was completed. Concurrently, the plans for the Full Scale Development program were prepared. The detailed results of both of these efforts are presented in Volume II of FR-19691-1.

Study efforts on this tripropellent configuration continued through the first extension of the Phase A contract. The results were given in Sections 2, 4 and 5 of FR-19691-2. Also during this period, vehicle studies produced information on bipropellant booster engines. P&W began work to define the characteristics of a bipropellant booster engine, and this work is documented in section 3 of FR-19691-2.

At this point in the conceptual design process, P&W's Manufacturing Division studies revealed innovative low-cost design concepts and manufacturing techniques for the STBE configuration.

The concurrent studies by the ALS vehicle contractors were now showing some results. These results showed that a bipropellant gas generator engine cycle is more cost effective than the tripropellant. Also, at about this same time, NASA changed the engine life requirement from 100 missions to 30 missions, a number thought to be more realistic. The tripropellant configuration was then set aside and the effort was focused upon the LO_2/methane bipropellant gas generator. This configuration had the second lowest life cycle cost after the original tripropellant selection in the evaluation. During the second extension of the Phase A contract, P&W completed its studies on the tripropellant engines and continued working on the design characteristics and configurations of several bipropellant engines, including the LO_2/methane gas generator engine. The results of these studies, including the work performed on the tripropellant engine during that extension period, is given in FR-19691-3. The reasons why methane was consistently better than either propane or RP-1 are given in Table 1-4.

Table 1-4. *Methane Advantages Over Propane and RP-1*

• Highest Combustion Efficiency	• Very Stable Combustion
• More Predictable Heat Flux	• A Good Coolant, with High Coking Temperatures
• Clean Gas Generator Gas	• Allows Transpiration Cooling
• Simplifies Injector Design	• Allows Coaxial Injection of Gaseous Fuel
• Self Purging Reduces Cleaning Requirements	• Improves the Injector Face Cooling
	• Reduces environmental impact of spills. The volatile, non-toxic gas readily disperses.

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The conceptual design of the LO_2/methane bipropellant gas generator engine is presented in detail in paragraph 4.1.1. The plans for its full-scale development have been prepared and are presented in Section 5.0.

In late 1987, in-house studies by Pratt & Whitney started to show that a split expander engine cycle would be more cost effective than a gas generator cycle. The split expander cycle differs from the standard expander cycle used in the RL10 engine by separating a portion of the fuel flow at the first-stage pump and directing that flow directly to the injector. The remainder of the fuel flow completes the standard expander cycle. The total heat "pickup" in the nozzle by this flow is approximately the same as a standard expander cycle. This flow cools the chamber and drives the turbines. Since flow and temperature trade proportionally in turbine power, the split expander low flowrate at the higher temperature will provide the same turbine power. The pump work will be reduced due to the reduction in flow through the second stage pump. This reduced power requirement provides the capability for a higher chamber pressure. Furthermore, the increase in fuel temperature at the turbine inlet ensures that gaseous fuel will be maintained (throughout the turbine) at high thrust level conditions. With the approval of NASA, the analysis and evaluation of the split expander engine cycle became part of this contract effort. The analysis of the split expander continued through the remainder of the contract at a lower level of effort than the gas generator cycle. The details of the early design analyses of the split expander cycle engines can be found in paragraph 4.2.

In 1988 the ALS Vehicle Contractor studies began to show some advantage to having a common engine for both the booster and the core vehicles, i.e., one engine that could meet both the requirements of the STME when operated with hydrogen/oxygen propellant, and the requirements of the STBE when operated with LO_2/methane propellants. The design analyses of such an engine, presented in paragraph 4.1.2, showed that the design was possible, but penalized the hydrogen/oxygen core vehicle engines because of the required additional weight.

At about this time it became more evident that the immediate need was for a LO_2/hydrogen engine (STME). The LO_2/methane engine requirement was slipping toward the end of the ALS program life.

The (STBE) contract emphasis then finally shifted to a LO_2/methane booster engine that could be obtained by modifying the STME engine design and still attain a sea level thrust of 600,000 lbf or greater. This requirement was met by both the gas generator engine cycle and the split expander engine cycle. This engine is known as a Derivative STBE.

Task IV (SOW Task 5.4) carried all of the selected engine configurations through the preliminary design phase. The trade studies and the resultant design details are presented in Section 4.0, Design Definition Document.

SECTION 2.0
ENGINE REQUIREMENTS AND CONFIGURATIONS

This section describes the work conducted under Task I (SOW Task 5.1). This task developed the requirements for the Space Transportation Booster Engine (STBE) and the performance and physical characteristics of the candidate engine configurations. The results of this study are presented in P&W Interim Report FR-19691-1.

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SECTION 3.0
CONFIGURATION EVALUATION AND SELECTION CRITERIA

This section describes the effort conducted under Tasks II and III (SOW Tasks 5.2 and 5.3). The results of this effort are presented in P&W Interim Report FR-19691-1.

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SECTION 4.0
DESIGN DEFINITION OF SELECTED ENGINE CONFIGURATIONS
(DESIGN DEFINITION DOCUMENT DR-8)

This section describes the work conducted under Task IV (SOW Task 5.4). This task developed the conceptual designs of the selected STBE engine configurations.

The selected STBE engine configurations include:

- Derivative LO_2/CH_4 Gas Generator Cycle Engine
- Unique LO_2/CH_4 Gas Generator Cycle Engine
- Common LO_2/CH_4 Gas Generator Cycle Engine
- Unique LO_2/RP-1 Gas Generator Cycle Engine
- Derivative LO_2/CH_4 Split Expander Cycle Engine
- Unique LO_2/CH_4 Split Expander Cycle Engine
- Unique LO_2/CH_4 Tap-Off Cycle Engine.

Included is a preliminary Interface Control Document (ICD) and preliminary Contract End Item specification.

4.1 GAS GENERATOR CYCLE ENGINES

4.1.1 Derivative STBE Gas Generator Cycle Engine

4.1.1.1 Engine Design Evolution

The Derivative STBE gas generator cycle engine concept began as a result of the common engine studies. In addition, the need for a unique engine, optimized for core vehicle use, ruled out the possibility of funding a separate, unique booster engine design as well. As discussed in paragraph 4.1.3.1, the common engine designs consisted of a common O_2/H_2 gas generator cycle engine that had slightly reduced performance characteristics than the unique STME and a 644K common LO_2/CH_4 Gas Generator Cycle engine that had reduced thrust compared to the 750K unique STBE. Although hardware commonality between the two engines was maximized, the concept proved to be unacceptable when the following ground rules were established:

1. No performance, cost, or weight penalties of the unique STME engine design are permitted

2. The STBE engine will use as much of the unique STME hardware as possible, and thus will be a derivative of the STME

3. The booster engine application must obtain 600K sea level thrust or greater.

The conceptual design that arose as a result of this study is known as the Derivative STBE; it is a derivative of the LO_2/LH_2 STME, but uses LO_2/CH_4 and is designed for booster applications. Figure 4.1.1.1-1 presents an engine assembly drawing and the overall engine characteristics. This derivative engine is the current baseline design for the STBE, therefore, the parametric equations, the ICDs, and CEI documents included in this report pertain to the derivative engine.

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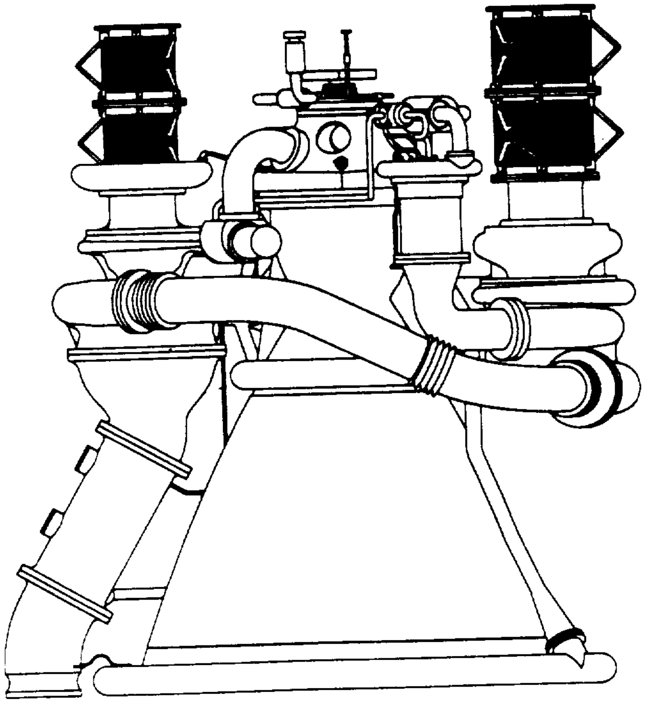

Propellants	CH_4/LO_2
Mixture Ratio	2.7
Chamber Pressure	2,250 psia
Thrust - Vacuum	711,823 lb
- Sea Level	644,898 lb
Specific Impulse - Vacuum	328.4 sec
- Sea Level	297.5 sec
Nozzle Area Ratio	28
Diameter	91 in.
Length	99 in.
Weight	6,960 lb

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Figure 4.1.1.1-1. STBE Derivative Gas Generator Engine Design Conditions

4.1.1.2 Engine Cycle

The STBE derivative is a LO_2/CH_4 gas generator cycle adapted from the STME LO_2/LH_2 gas generator (GG) cycle engine. The STBE operates at a main chamber pressure of 2250 psia with a sea level thrust of 645K lbf. The nozzle area ratio for this engine is 28:1 and delivers a sea level specific impulse of 297.5 seconds. Figure 4.1.1.1-1 presents selected engine characteristics at the fixed operating conditions. Table 4.1.1.2-1 gives parametric equations for the vacuum I_{sp}, nozzle exit diameter, and total engine length.

Table 4.1.1.2-1. *STBE Derivative Gas Generator Parametrics*

Vacuum I_{sp}	=	$-319.7 - 182.2\,(OF) - 130.6\,\frac{OF}{AE} + 689.6\sqrt{OF} + 1.638\sqrt{AE} + 0.0048\,(P_c)$
Nozzle Exit Diameter (in.)	=	$88.00\left(\frac{F_{vac}}{711823}\right)^{0.5}\left(\frac{AE}{28}\right)^{0.5}\left(\frac{2250}{P_c}\right)^{0.5}$
Total Engine Length (in.)	=	$69.6\left(\frac{Dia}{88.0}\right)^{1.14}\left(\frac{711823}{F_{vac}}\right)^{0.07}\left(\frac{28}{AE}\right)^{-0.06} + 0.0000302\,(F_{vac}) + 12.0$

Where:

			Validity Range
OF	=	Inlet Mixture Ratio	2.3 to 3.7
AE	=	Nozzle Area Ratio	20 to 50
P_c	=	Chamber Pressure	1400 to 2400
F_{vac}	=	Vacuum Thrust (lb)	200,000 to 1,000,000

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Components of the STBE derivative that will be common with the STME are the main injector, gas generator, tubular nozzle, engine controller, igniters, GO_2 HEX, POGO suppressor,

instrumentation, vehicle interfaces, and 80 percent of the ducting. Items that will be redesigned for the STBE derivative are the combustion chamber, oxidizer pump, oxidizer turbine, fuel turbine, GG oxidizer valve, GG fuel valve, and the gimbal. Table 4.1.1.2-2 summarizes the common hardware components between the STME and Derivative STBE gas generator engines.

Table 4.1.1.2-2. *STME and Derivative STBE Gas Generator Engines — Common Hardware Components*

Turbomachinery	*Combustion Devices*
• Fuel Pump Housing Flow Paths	• Gas Generator Injector Interpropellant Plate
• Fuel Pump Impeller Flow Path	• Gas Generator Injector Housing
• Ball and Roller Bearings	• Gas Generator Combustion Chamber
• Turbine Outer Seals	• Gas Generator Combustion Chamber Liner
• Tiebolt Shaft and Disks (Modified Blade Attachments)	• Tubular Nozzle
• Internal Labyrinth Seals	• Nozzle Inlet Manifold
• Major Flange Seals	• Nozzle Discharge Manifold
• Bolts, Nuts, Studs, Washers, Pins	• Main Injector Interpropellant Plate
• 1st and 2nd-Stage Impeller Castings	• Main Injector Housing
• Uniform Cross Section Static Housing Seals	• Main Injector Faceplate
• Inducer Retaining Bolts	• Igniter Assembly — Main Injector
• Blade Retaining Rings, Tip Seals	• Igniter Assembly — Gas Generator Main Chamber to Injector Flange, Seals, Fasteners
• Spacers, Bearing Sleeves, Wave Washers Made from Same Forging or Identical Hardware	
Engine Controls	*Engine Assembly*
• Engine Controller	• Ducting 80% Small Lines 80% Large Lines
• Engine and Component Instrumentation	• GO_2 HEX
	• POGO Suppressor
	• Fuel Inlet Flex Joints
	• Fasteners, Seals

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4.1.1.2.1 Flowpath Description

A simplified flow schematic for the STBE derivative is presented in Figure 4.1.1.2-1 showing the major components and flowpaths. Liquid methane and liquid oxygen enter the engine at a NPSH level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps, with no boost pumps required.

The two-stage methane pump operates at 10673 rpm to deliver fuel at the required pressure of 4621 psia. From the pump exit the fuel flows through the fuel shutoff valve (FSOV) and to the chamber/nozzle cooling jacket manifold where the flow splits so that 25 percent goes to the regenerative nozzle cooling jacket and 75 percent goes to the regeneratively cooled chamber jacket. The nozzle cooling flow is used entirely to fuel the gas generator while the chamber coolant flow is discharged at 409 R directly into the main chamber injector.

The high-pressure oxidizer pump operates at 7601 rpm to provide the oxygen pressure level of 3338 psia required by the cycle. From the pump exit, approximately three percent of the LO_2 flow is diverted to the gas generator oxidizer control valve and subsequently to the gas generator. The bulk of the LO_2 flow (97 percent) flows through the main oxidizer control valve and directly to the main chamber injector.

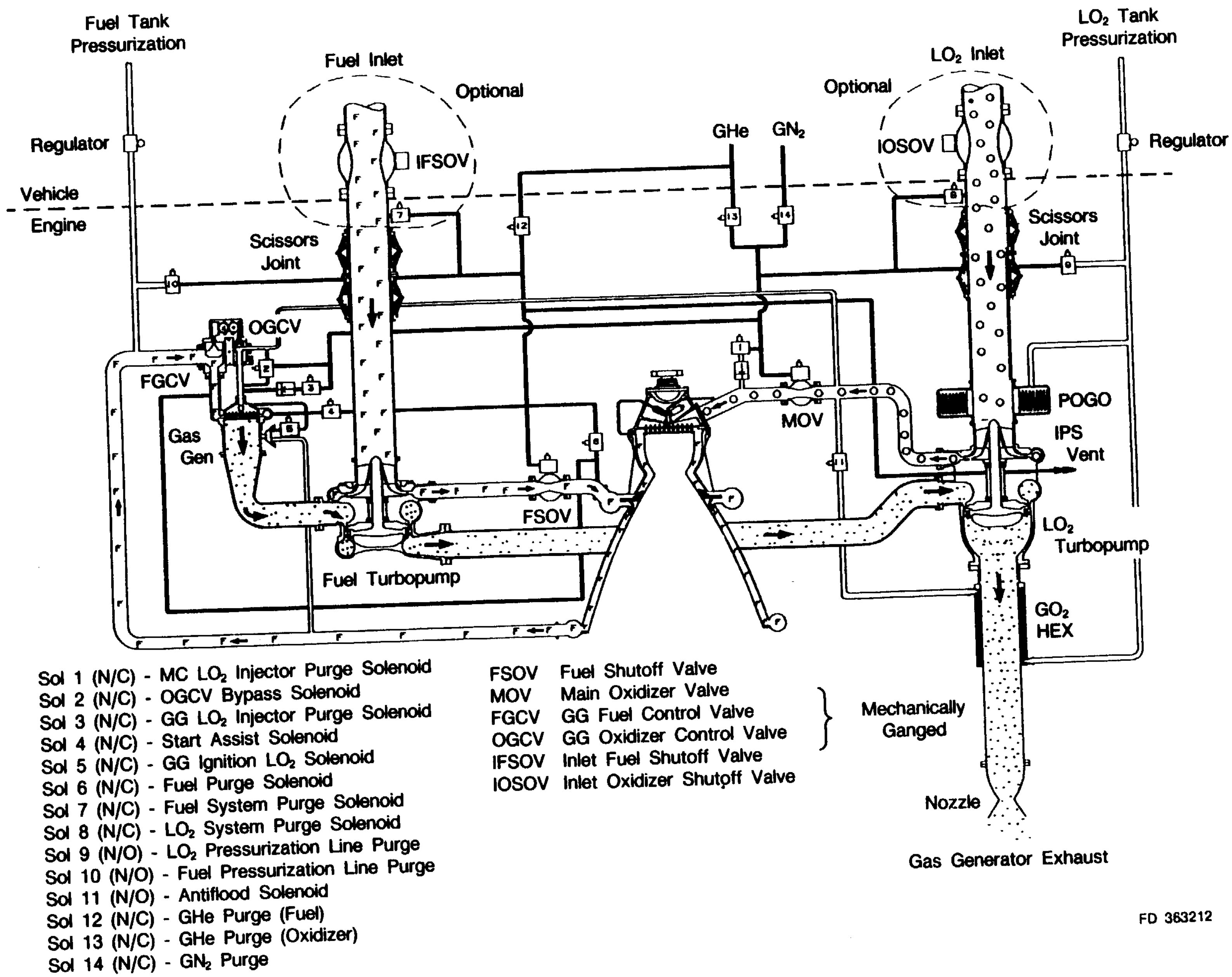


Figure 4.1.1.2-1. Simplified Flow Schematic for STBE Derivative Gas Generator Cycle Engine

The high-pressure, high-temperature (1688 psia/1800 R) gas from the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas flow is initially expanded through the methane turbine and is subsequently routed to a second turbine which powers the oxidizer pump. The turbine exhaust gas is then diverted through the gaseous oxygen heat exchanger (for tank pressurization) and then discharged through a nozzle of area ratio 5.0 to produce thrust.

4.1.1.2.2 Engine Operation

The engine will be preconditioned using liquid flow from the tanks to soak the turbopumps until they are sufficiently cooled. The inlet valves will be opened, allowing liquid from the tanks to flow down to the turbopumps and letting any vapors to percolate back up to the tanks to be vented.

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor or on the pad, because all fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead start sequence, the GG LO_2 injector is primed prior to opening the GG fuel valve to assure liquid oxidizer flow, thus eliminating turbine temperature spikes due to oxidizer phase change. After the GG LO_2 valve is opened, the main oxidizer valve (MOV) is opened followed by both the fuel GG valve and the fuel shutoff valve (FSOV). Helium spin assist is provided to the gas generator to help start the turbopump rotating and is discontinued early in the engine acceleration. Gas generator and main chamber ignition is accomplished with common design dual electrical spark-excited, oxygen/methane torch igniters. Engine acceleration is accomplished by open-loop scheduling of the gas generator oxidizer control valve.

Main stage thrust control is provided through open loop control of the GG oxidizer valve. Engine mixture ratio is preset by trim of the main oxidizer valve.

Engine shutdown is accomplished using a time based scheduling of the propellant valves. The gas generator oxidizer valve is closed first to power down the turbopumps, then the GG fuel valve is shut along with the MOV. The FSOV is closed when the pump is at low rpm. Provisions for post shutdown purging of propellants is provided.

4.1.1.3 Turbomachinery

4.1.1.3.1 Oxidizer Turbopump Hardware Description

The oxidizer turbopump is configured as a single-stage centrifugal shrouded impeller pump with an inlet inducer driven by a two-stage axial flow turbine. The design features of this turbopump are shown in Figure 4.1.1.3-1. The inducer and impeller, made of fine grained cast and Hot Isostatically Pressed (HIP) Inconel 718, is coupled to the turbine through a single turbine disk with an integral shaft made of Waspaloy. Pump and turbine inlet and discharge housings are fabricated from fine grained cast and HIP Inconel 718 to minimize machining costs. Turbine blades and vanes are made from cast Mar-M-247 nickel base alloy. The ball and roller bearings, made of 440C material will be used to support the pump rotor system. Investigations are ongoing to find an alternate cyrogenic bearing material or combination. Any data and/or technology that is obtained through this investigation or the Space Shuttle Main Engine Alternate Turbopump Development (SSME-ATD) program, will be applied to the STME and STBE turbopump bearings and bearing systems.

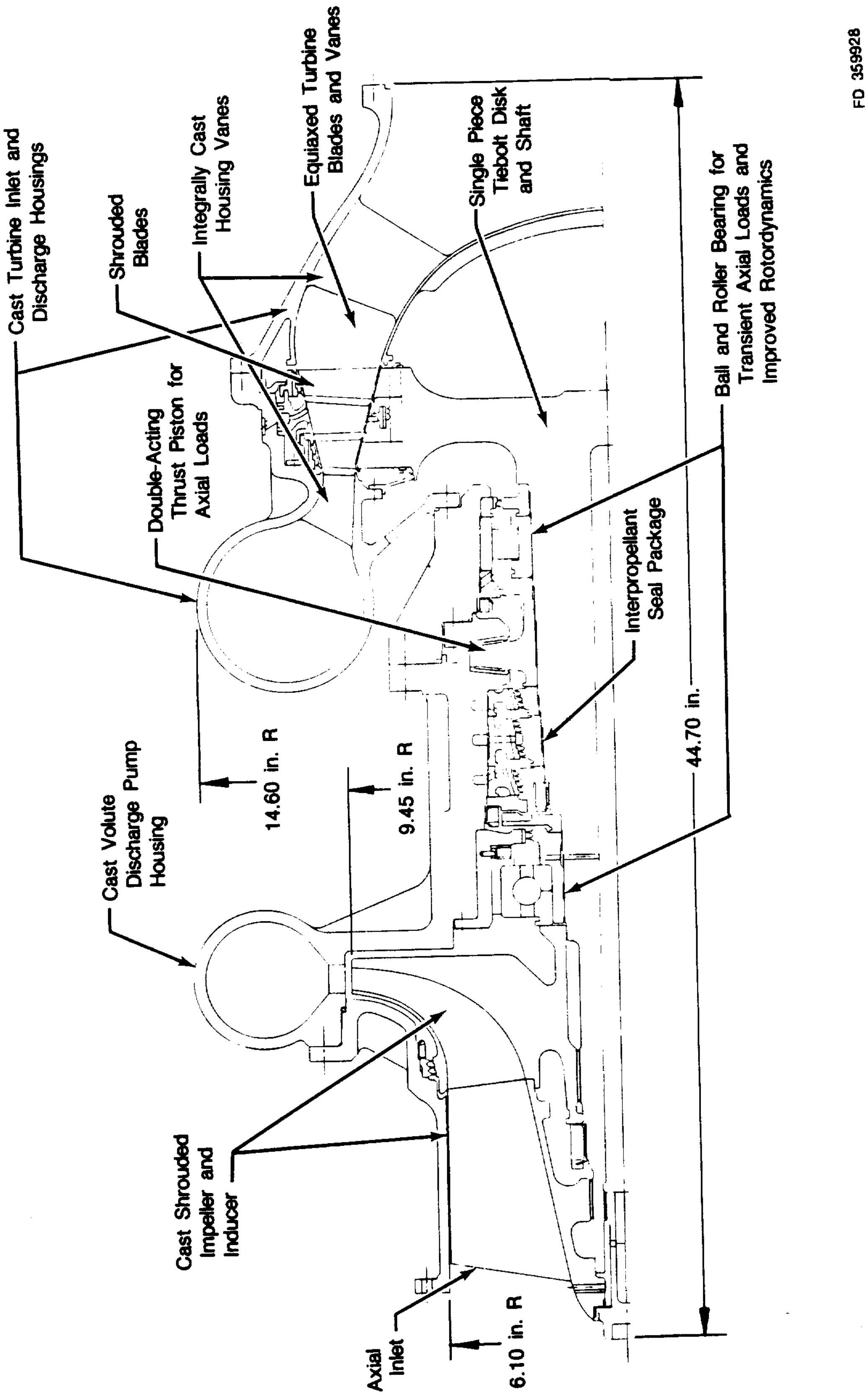


Figure 4.1.1.3-1. STBE Derivative Gas Generator LO_2 Turbopump

The rotor thrust balance system is accomplished through the incorporation of a double acting thrust balance system on the turbine side of the interpropellant seal turbopump in a liquid fuel environment so as to avoid any rub in a LO_2 environment. Externally supplied high pressure fuel (methane) is used for thrust piston actuation and for roller bearing and turbine coolant. The rotating thrust piston is made of forged Inconel 718 and its mating surface of the stationary housing is an insert made of Bearium B-10 material (leaded bronze). Axial travel of the rotor is controlled at this location.

The double-acting thrust piston provides thrust balance capability to the rotor system by controlling axial imbalance loads during startup, steady-state, and shutdown operation. As an axial imbalance load occurs, the rotor moves axially, which opens an orifice that supplies high-pressure fuel to the side of the piston in which the rotor has traveled. At the same time, the opposite piston face is now vented to low pressure fuel, resulting in a reaction thrust load that restores the rotor to its initial position.

While the roller bearing is cooled by fuel, the ball bearing is cooled with LO_2. Oxidizer flow along the backside of the impeller is used as bearing coolant, then is recirculated to the inducer inlet through a controlling orifice/hole in bearing carrier and shaft. Bearing DNs for the ball and roller bearings is 0.88×10^6 and 1.06×10^6 respectively. In addition, damper seals will be used to assist in rotor damping.

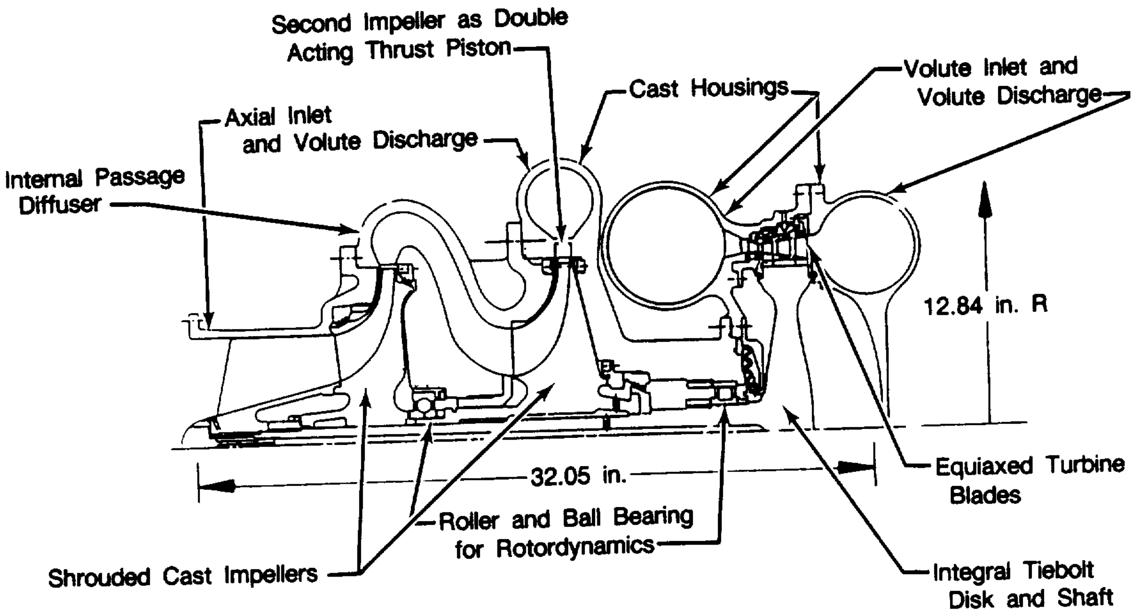
The interpropellant seal package employs a labyrinth seal design with a helium buffer cavity. This design is similar to the SSME ATD LO_2 turbopump design. An oxidizer-side vaporizer is incorporated to reduce the overboard leakage.

The turbine inlet housing is a cast volute integrating the first-stage turbine vane, and contains the placement of the turbine tip seal lands. A gas-cooled liner is not required at this location because of relatively low temperatures and pressures as compared to the fuel turbopump turbine inlet. Attachment of the inlet housing to the pump housing is achieved with a flexible arm designed to provide thermal compatibility between the two housings.

The turbine discharge housing is a fine grained cast and HIP Haynes 230 configuration which incorporates an exit guide vane. This guide vane is required, due to the relatively high second-blade exit angle, to avoid excessive flow losses resulting from redirecting the flow into an axial direction.

4.1.1.3.2 Fuel Turbopump Hardware Description

The fuel turbopump is configured as a two-stage centrifugal shrouded impeller pump with an inlet inducer driven by a two-stage axial flow turbine The design features of this turbopump are shown in Figure 4.1.1.3-2. The inducer and impeller, made of fine grain cast titanium A-110 ELI, are coupled to the turbine through an integral turbine disk shaft made of forged Waspaloy. Pump and turbine inlet and discharge housings are fabricated from fine grained cast and HIP Inconel 718 to minimize machining costs. Turbine blades and vanes are made from cast Mar-M-247 nickel base alloy. The ball and roller bearings, made of 440C material, will be used to support the turbopump rotor system. In addition, damper seals will be used to assist in rotor damping. These fluid hydrostatic bearings are supplied with leakage flows from the impeller back face.



Materials

Pump Impeller	Titanium
Turbine Disk/Shaft	Super A-286
Turbine Blades	MAR-M-247
Turbine Vanes	MAR-M-247
Housings	INCO 718, MAR-M-247 and Haynes 230

FD 366121

Figure 4.1.1.3-2. STBE Derivative Gas Generator Fuel Turbopump

The rotor thrust balance system is accomplished by incorporating the thrust piston into the second-stage impeller. A double acting, double orifice thrust piston has been configured into the front and back side of the impeller. The thrust piston is designed to control axial imbalance loads during engine startup, steady-state, and shutdown conditions. As the thrust imbalance load occurs, the rotor moves axially, which then opens an orifice at the impeller tip, introduces pump discharge high pressure fuel to the side of the impeller in which the rotor has traveled. At the same time, the opposite impeller face is vented to low pressure fuel, resulting in a reaction thrust load to restore the rotor axial position. Both sides of the thrust piston are fed with second-stage impeller discharge pressure. Axial travel is limited by a forward stop on the impeller ID shroud face and by an aft stop on the ID of the impeller back face.

The ball and roller bearings are the same bearings used on the SSME-ATD fuel turbopump. The ball bearing is cooled by first-stage discharge pressure bled off the impeller back face and flow controlled by the labyrinth seals near the outer diameter of the impeller. The roller bearing is cooled by second-stage discharge pressure that is supplied to the bearing via internal passages through the pump housing. Roller bearing coolant is then discharged into the turbine disk cavity to be used as turbine coolant. Bearing DNs for the ball and roller are 0.64×10^6 and 0.78×10^6 respectively.

The turbine inlet and discharge housings are fine grain and HIP casting Haynes 230 volutes. Attachment of the inlet housing to the pump housing is achieved with a thermally compatible designed flexible arm.

A diaphragm type lift-off seal (similar to the ATD fuel turbopump) is incorporated in the fuel pump design to prevent cooldown flow from entering the turbine during the pre-start sequence. At engine start, pump pressure increases so that lift-off seal is deflected and flow is permitted through the bearing and into the turbine for additional turbine cooling requirements.

4.1.1.3.3 Turbomachinery Rotordynamics

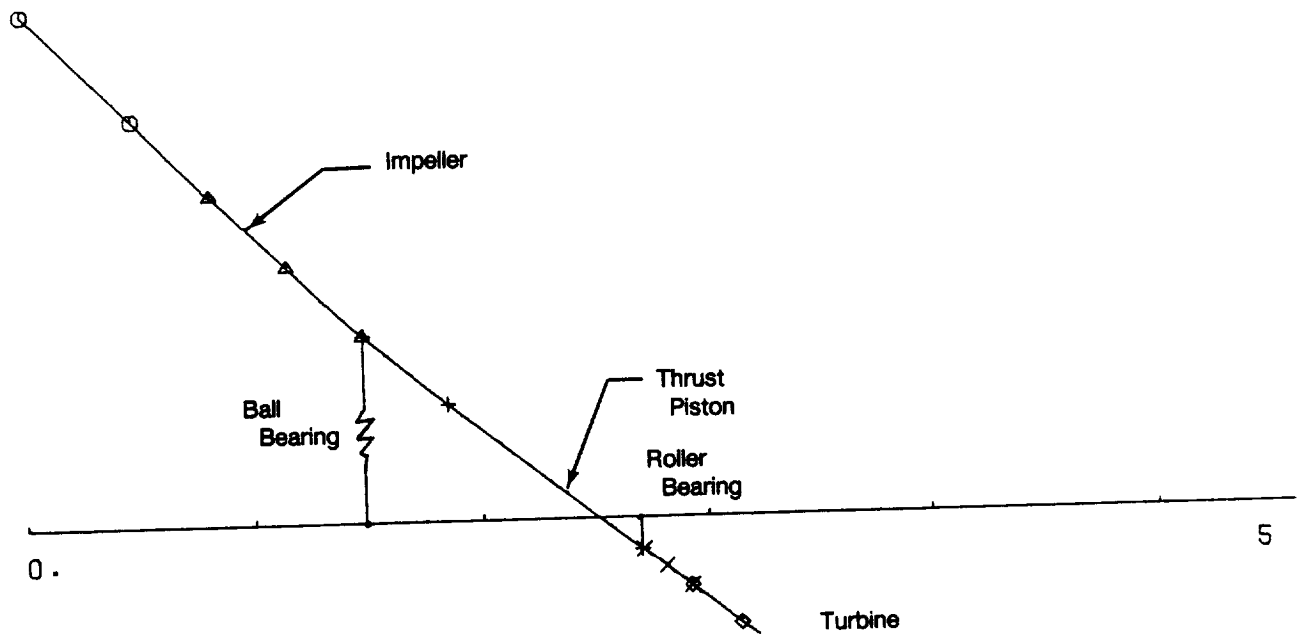
The P&W Advanced Launch Systems (ALS) Program is designed to produce reliable, low-cost rocket engine turbopumps. Pratt & Whitney uses proven design criteria and analytical methods in the design of rotordynamic operation for jet engine rotors and rocket engine turbopumps. Each Derivative Gas Generator Oxygen Turbopump (DGGOT) and Derivative Gas Generator Fuel Turbopump (DGGFT) design incorporates configuration changes which result in stiffer rotors, bearings, and rotor support structures with the addition of roughened stator damper seals. For optimum rotordynamics, each rotor is supported by strategically located, stiff, durable bearings. These changes result in a significant improvement to the first fundamental bending mode of the rotor, moving it well beyond the maximum operating design speed. This, in addition to an improved rotor balance procedure, results in an effective low speed balance of the rotor for low synchronous response. Rotor stability in the DGGOT and the DGGFT have been improved by designing the turbopumps to operate below the first vibrational mode of the rotor. Increased stability margin in each turbopump is provided by the introduction of roughened stator damper seals into the design.

4.1.1.3.3.1 Oxidizer Turbopump Rotordynamics

The primary goals in the design of the DGGOT have been to provide: (1) greater than 20 percent margin over design speed for the first fundamental bending mode for low synchronous response, (2) a sufficient stability margin, and (3) a high integrity rotor balance. Meeting these provisions has required optimization of the mechanical design of the rotor, bearings, rotor support, damper seals, and housings for successful rotordynamic characteristics. The initial P&W DGGOT design moved the first fundamental rotor bending mode, with high strain energy, to well above the design speed, effectively eliminating the synchronous response due to rotor imbalance. The pitch and bounce modes of the rotor occur at 99 and 199 percent of operating speed. These modes are classified as rigid body modes and are of relatively low rotor strain energy content. They are shown in Figures 4.1.1.3-3 and -4.

Rotor bearing stiffness plays an important role in the dynamic behavior of all turbomachinery. In high-pressure rocket turbopump designs, P&W realizes the need for the combined rotor support system (i.e., bearing, carrier, and backup structure) to approach or exceed the relative stiffness of the rotor structure.

The rotor critical speed analysis has been used to set initial design requirements for each bearing stiffness. The pump end bearing is a large diameter, high load capacity ball bearing with minimal internal radial clearance (IRC) and deadband. The turbine end bearing is a large diameter, high load capacity roller bearing with a negative IRC. High rotor stiffness, coupled with stiff rotor design, without exceeding successful P&W bearing DN experience, ensure that successful rotordynamics criteria are met for this application.



FD 363180

Figure 4.1.1.3-3. STBE Derivative Gas Generator LO_2 Turbopump Critical Speeds Analysis Showing the Pump Pitch Mode of the Rotor at 99% Design Speed (RPM = 7016)

Rotordynamic stability analysis will be used as a design tool to determine the final damper seal configuration requirements for optimized system dynamics. However, the DGGOT is designed such that damper seals are not critical to the dynamics of the rotor system. Each of the seals is designed for high damping, moderate stiffness, and minimal leakage. The incorporation of damper seals into the turbopump design provides: (1) reduced synchronous response throughout the operating speed range resulting in lower dynamic bearing loads and rotor deflections, (2) increased margin on the onset speed of instability (OSI), and (3) additional rotor load support between the bearings. Locations for the damper seals are being investigated to be incorporated into the next phase of design.

The DGGOT design provides for improved rotor balancing. Each major rotating component will be double piloted and indexed to the through tiebolt for positive concentricity control and balance repeatability. In addition, each major rotating component will be dynamic check balanced in detail to provide minimal residual force and moment imbalance. The dynamic balance of the rotor assembly will be completed with corrections in two planes.

The DGGOT design proposed by P&W results in acceptable rotordynamic characteristics throughout the operating range. The lightweight stiff rotor and bearing design has reduced the synchronous response due to the first fundamental rotor bending mode having been driven to more than 640 percent above the design speed. Thus, the stability of the rotor is significantly improved by avoiding the subsynchronous excitation associated with the critical speeds below 50 percent of the design speed.

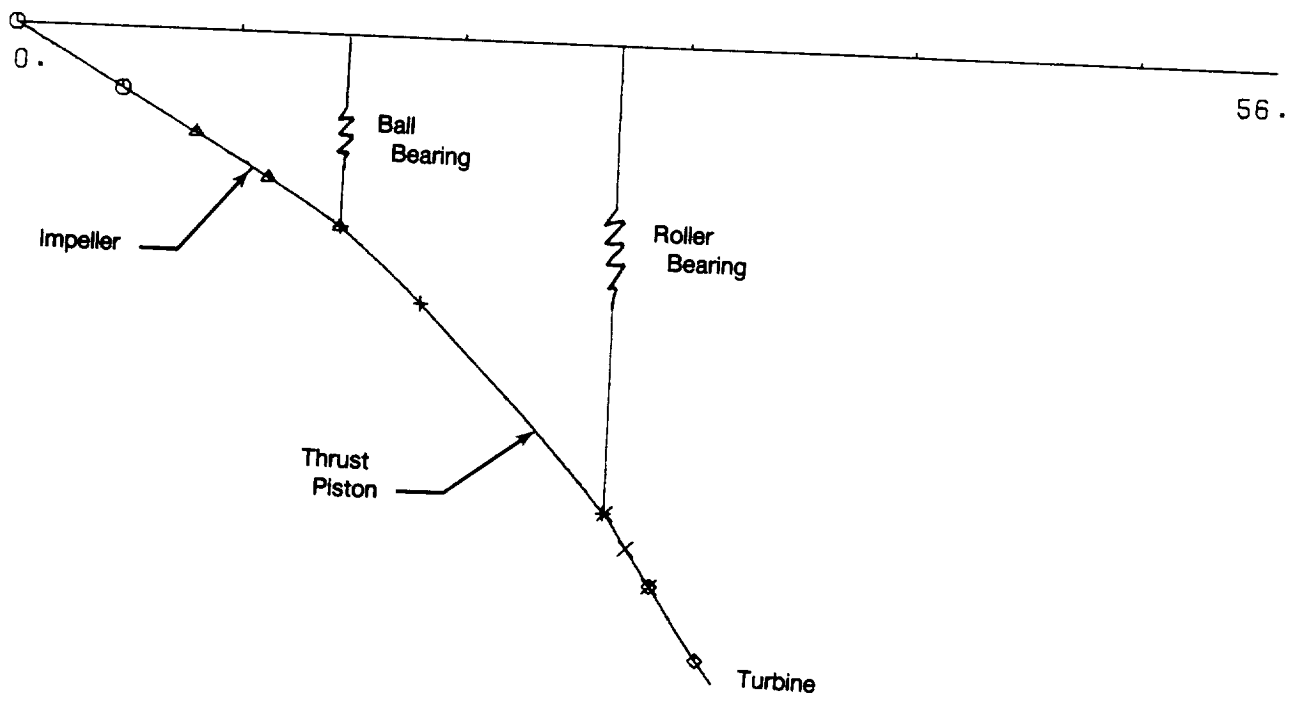


Figure 4.1.1.3-4. STBE Derivative Gas Generator LO_2 Turbopump Critical Speeds Analysis Showing the Turbine Bounce Mode of the Rotor at 199% Design Speed (RPM = 14556)

A critical speed summary for the DGGOT is provided below.

W_{cr} *(rpm)*	*% Design Speed*	*% Rotor Strain Energy*	*Mode Description*
7016	99	7.3	Pump Pitch
14556	199	6.4	Turbine Bounce
54067	740	95.5	1st Bending

4.1.1.3.3.2 *Fuel Turbopump Rotordynamics*

The three primary goals in the design of the DGGFT have been to provide: (1) a subcritical rotor design, (2) a sufficient stability margin, and (3) a high integrity rotor balance. Meeting these provisions has required optimization of the mechanical design of the rotor, bearings, rotor supports, damper seals, and housing for successful rotordynamic characteristics.

The initial phase of the DGGFT design has moved the fundamental rotor bending mode (with high rotor strain energy) to 149 percent above the design speed. Consequently, the synchronous resonant response due to the rotor imbalance is almost completely eliminated. The two modes occurring at 110 and 154 percent of the operating speed are low rotor strain energy rigid body modes of the rotor, as shown in Figures 4.1.1.3-5 and -6, respectively.

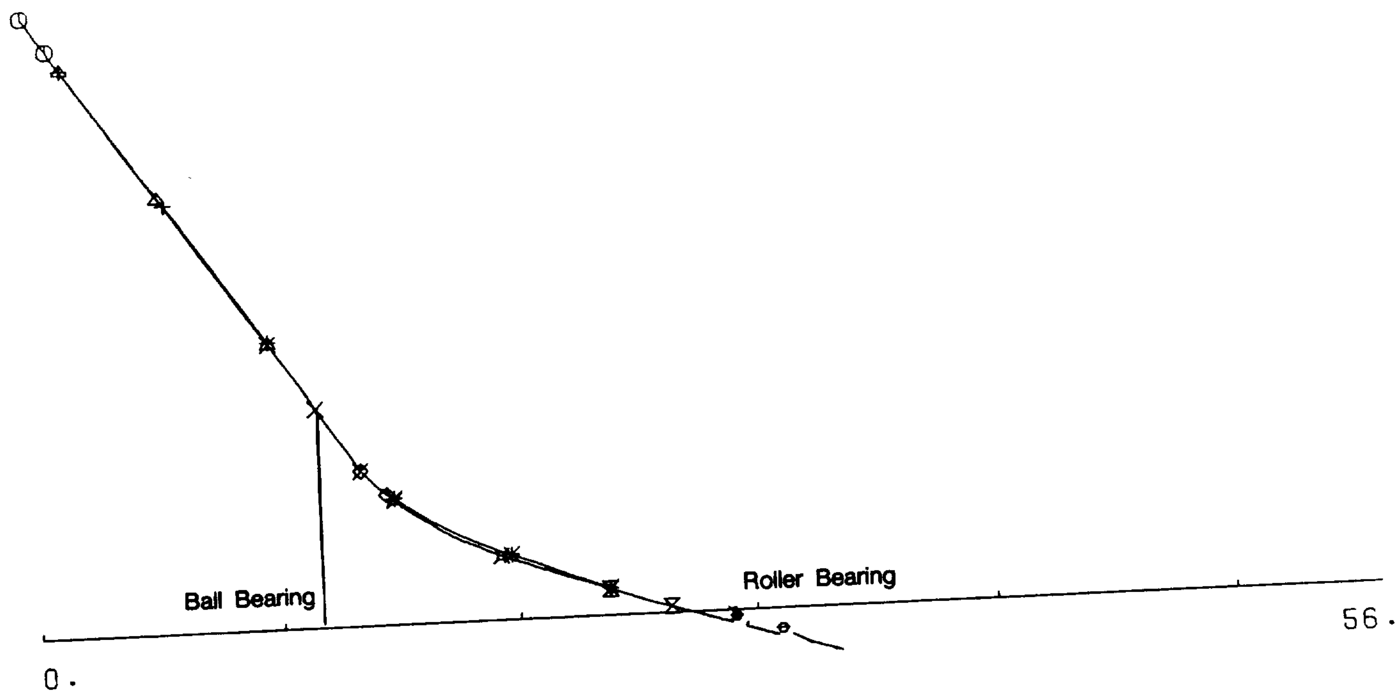


Figure 4.1.1.3-5. STBE Derivative Gas Generator Fuel Turbopump Critical Speeds Analysis Showing the Pump Bounce-Turbine Pitch Mode of the Rotor at 110% Design Speed (RPM = 11829)

Rotor bearing stiffness plays a very important part in the dynamic behavior of all turbomachinery. In high-pressure rocket turbopump designs, P&W realizes the need for the combined rotor support system stiffness (bearing, carrier, and backup structure) to approach or exceed the relative stiffness of the rotor structure to minimize rotor strain energy.

The rotor critical speed analysis has been used to set initial design requirements for each bearing stiffness. The pump end bearing is a large diameter, high load capacity ball bearing with minimum IRC and deadband. The turbine end bearing is a large diameter, high load capacity roller bearing with negative IRC. Without exceeding successful P&W bearing DN experience, high rotor support stiffness, coupled with a stiff rotor design in this application, ensure that successful rotordynamic design criteria are met.

Rotordynamic stability is further improved by eliminating the subsynchronous rotor excitation associated with the first rotor mode below 50 percent of the design speed. This has been accomplished by moving the first rotor rigid mode to 110 percent of the design speed and by the use of roughened stator damper seals at the impeller interstage locations.

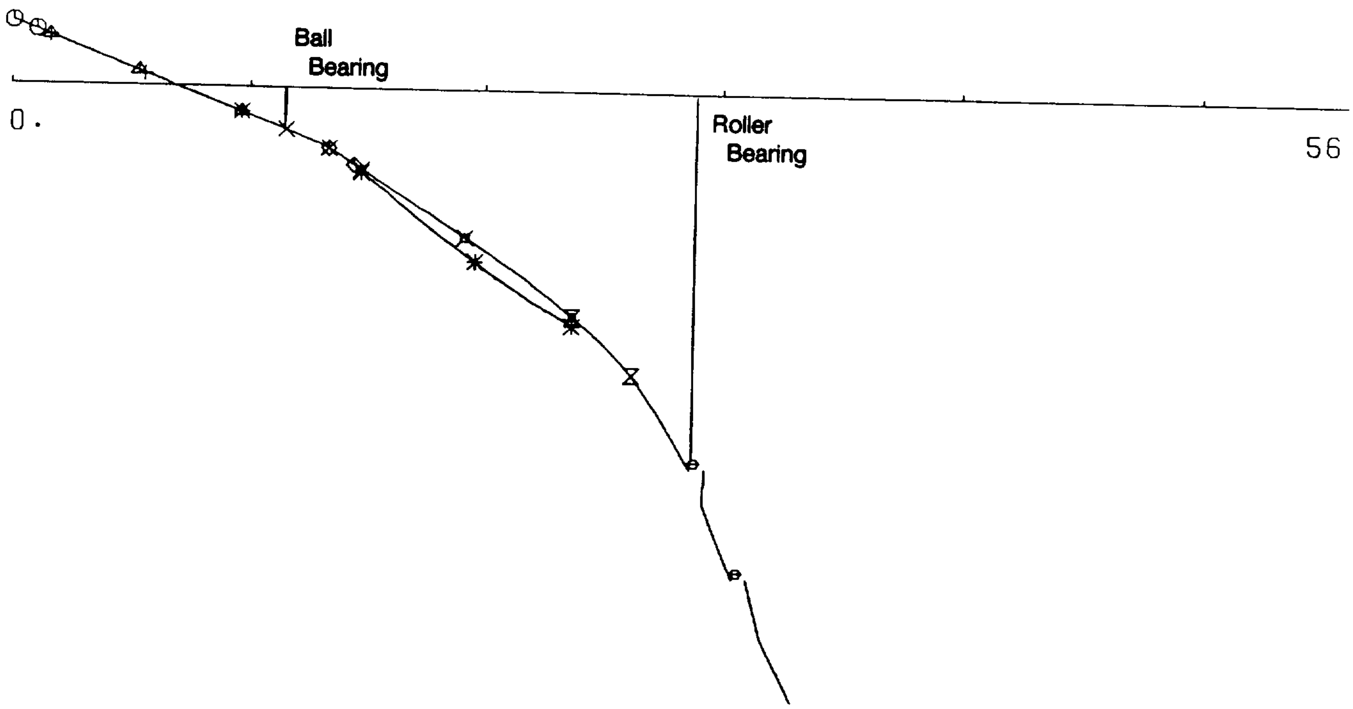


Figure 4.1.1.3-6. STBE Derivative Gas Generator Fuel Turbopump Critical Speeds Analysis Showing the Pump Pitch-Turbine Bounce Mode of the Rotor at 154% Design Speed (RPM = 16509)

Rotordynamic stability analysis will be used as a design tool to determine the final damper seal configuration requirements for optimized system dynamics. However, the DGGFT is designed such that damper seals are not critical to the dynamics of the rotor system. Each of the seals is designed for high damping, moderate stiffness, and minimal leakage. The incorporation of damper seals in the turbopump provides reduced synchronous response throughout the operating range resulting in low dynamic bearing loads and rotor deflections, sufficient margin on the OSI, and additional rotor load support. Parametric studies on the interstage damper seal locations will be presented in Phase B design.

Balance provisions and techniques used for DGGFT are identical to those used for the DGGOT.

A critical speed summary for DGGFT is provided below.

W_{cr} (rpm)	% Design Speed	% Rotor Strain Energy	Mode Description
11829	110	22.2	Pump Bounce-Turbine Pitch
16509	154	9.8	Pump Pitch-Turbine Bounce
26671	249	82.0	1st Bending

4.1.1.4 Combustor

The most commonly used models of steady-state rocket combustion are based on the assumption that droplet vaporization is the controlling mechanism in establishing combustion residence time. While these models have proven highly useful in combustion at moderate chamber pressure (100 to 1000 psi), the models break down at the conditions of interest for this program. The primary reason for this is that when a droplet is heated above its critical temperature and the total pressure of the environment is above the droplet critical pressure, boiling cannot occur. Under these conditions the small pockets of propellant, which technically are no longer droplets, must be dispersed through diffusion or forced convection. Both the rate of dispersion and the rate-of-change of dispersion as a function of temperature are different than for subcritical vaporization.

Pratt & Whitney has developed a rocket combustion model directly aimed at the high pressure conditions planned for this effort. The basic elements of the model are droplet formation, drop heating, ignition delay and drop burning. A droplet mass mean diameter (MMD) and distribution based on the predicted performance of a tangential entry swirl coaxial injector element is used as input for the model. The model calculates the droplet heating time based on a method proposed by Wieber (Reference 1) and the ignition delay time is based on the P&W correlation for methane. The percentage of the droplet distribution burned is calculated by a method proposed by Rosner (Reference 2). In the Rosner model, the combustion rate is controlled by diffusion once the liquid droplet reaches critical conditions, i.e., when the droplet reaches its critical temperature and the combustion process pressure is above the critical pressure of the fluid. Under these conditions, increasing the pressure increases the density of the diffusing reactants which reduces the rate of diffusion and therefore reduces the rate of combustion. It is currently planned to check the efficiency predictions of the combustion model and the ignition delay correlation with test work now in progress at UTRC. The model results for the derivative STBE engine are shown in Figure 4.1.1.4-1 and indicate a L* of 34 inches is required to achieve the 98 percent C* efficiency specified for the engine.

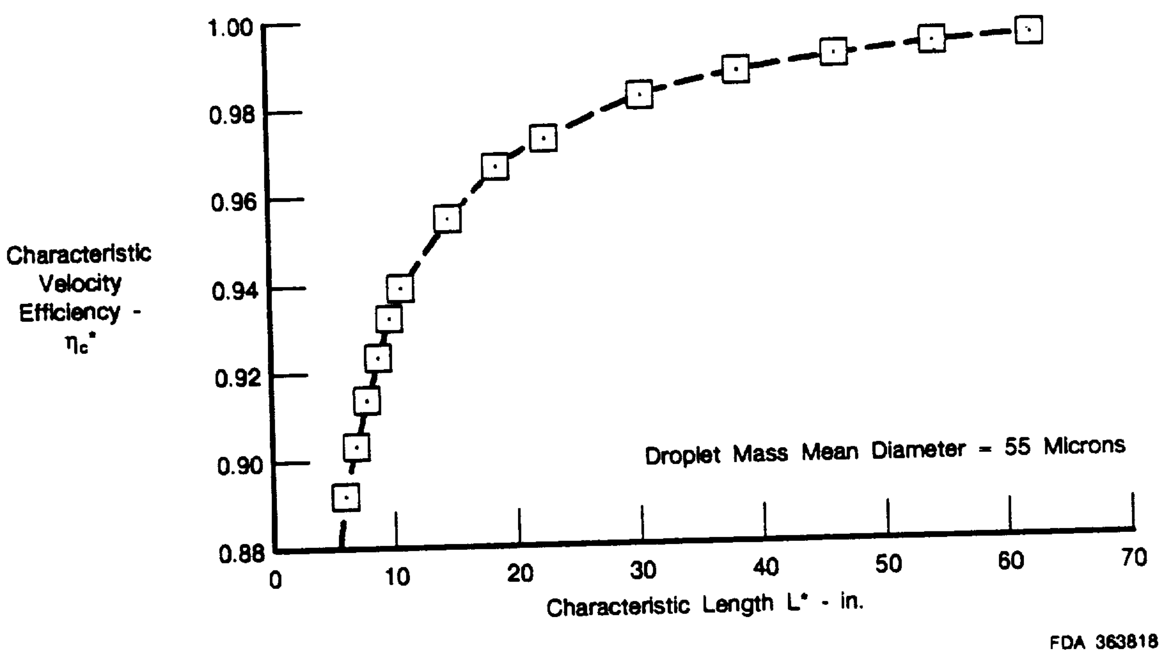


Figure 4.1.1.4-1. Efficiency vs Length for STBE Derivative Gas Generator Combustion Chamber

Injector Elements

The injector element performance is critical to the combustion efficiency and stability of the combustion process. Two important parameters relating to the injector performance and design are pressure drop and the number density of elements on the injector face.

The ΔP across the elements must be high enough to prevent "chug" or fuel system coupled instability (minimum six percent P_c for fuel and four percent P_c for LO_2). The ΔP is also important to the drop size distribution produced by the element and hence the combustion efficiency of the chamber. The element density sets the overall dimensions of the coaxial injection elements which must stay within manufacturing and contamination limits. In addition to these considerations, the derived STBE engine needed to be designed using the same injection elements as used in the STME unique chamber design. The actual design set for the gas generator cycle STME and derived STBE injector is given in Table 4.1.1.4-1. This injector meets the above design constraints in both engines. The injector is estimated to produce a LO_2 droplet spray with a 55 micron MMD in the STBE engine based on coaxial injector performance data recently taken for the National Aerospace Plane program by Pratt & Whitney. The main penalty involved in using the same injector elements in both the STME and derivative STBE is that a higher pressure drop is required in the STBE than would otherwise be required for injector element designs for that engine alone.

Table 4.1.1.4-1. *STME/Derivative STBE Injector Elements Dimensions and Operating Conditions*

	Derivative STBE	*STME*
Chamber Pressure-psi	2250	2250
Fuel Flow-lb/sec	442.3	164.7
ΔP Fuel-psi	293.0	170.0
LO_2 Flow-lb/sec	1541.0	1112.0
ΔP LO_2 psi	302.0	157.0
No. of Elements	890	890
Spud ID-in.	0.272	0.272
Annular Gap-in.	0.02	0.02

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Acoustic Liner

The derivative STBE combustor chamber will be provided with an acoustic liner to suppress combustion instability. The liner consists of a perforated surface that absorbs a portion of a reflected pressure wave, thereby damping the intensity of the reflected wave and decoupling the wave from the combustion process. A common measure of liner performance is the absorption coefficient which is equal to the energy absorbed divided by the incident wave energy. The absorption for a given liner design and operating conditions can be calculated by the P&W acoustic liner design deck (5612). This deck uses well established mathematical models of the acoustic liner operation along with empirical correlations of the acoustic aperture effective length and resistance for high sound levels. The predictions of this program were recently verified in work done under AFAL contract F04611-86-C-0115.

The acoustic liner design proposed for the derivative STBE core is listed in Table 4.1.1.4-2. To arrive at this design some of the parameters, such as the acoustic aperture hole diameter and length, had to be estimated. These parameters are usually set by the cooling channel dimensions and have a relatively small impact on acoustic absorption. The 0.05 area ratio (acoustic hole

area/total acoustic liner area) was set based on past parametric studies which have shown this value to be close to optimum. The backing cavity depth was set to maximize absorption at the first tangential frequency of the combustion chamber (1423 Hz). Experience has shown that this is the most likely frequency of combustion instability. The liner placement in the chamber (near the injector face) and length are based on experimental testing and design experience which has shown that combustion can be stabilized by ¼-length liners with a minimum of 20 percent acoustic absorption at the frequencies of interest. Further experimental verification of the acoustic liner design procedures and assumptions such as backing cavity gas temperature will be obtained in the planned testing of the STBE subscale test chamber under contract NAS8-37490. The predicted acoustic absorption of the STBE acoustic liner or a function of frequency is shown in Figure 4.1.1.4-2. The curve shows good acoustic absorption over a broad frequency range.

Table 4.1.1.4-2. STBE Derivative Engine Acoustic Liner Design and Operating Conditions

Chamber Pressure-psi	2250.0
Aperture — Gas Temperature-°R	2000
Aperture — Gas Molecular Wt.	22.2
Hole Diameter-in.	0.1
Hole Length-in.	0.35
Area Ratio	0.05
Backing Cavity Depth-in.	0.60
Liner Length-in.	4.8

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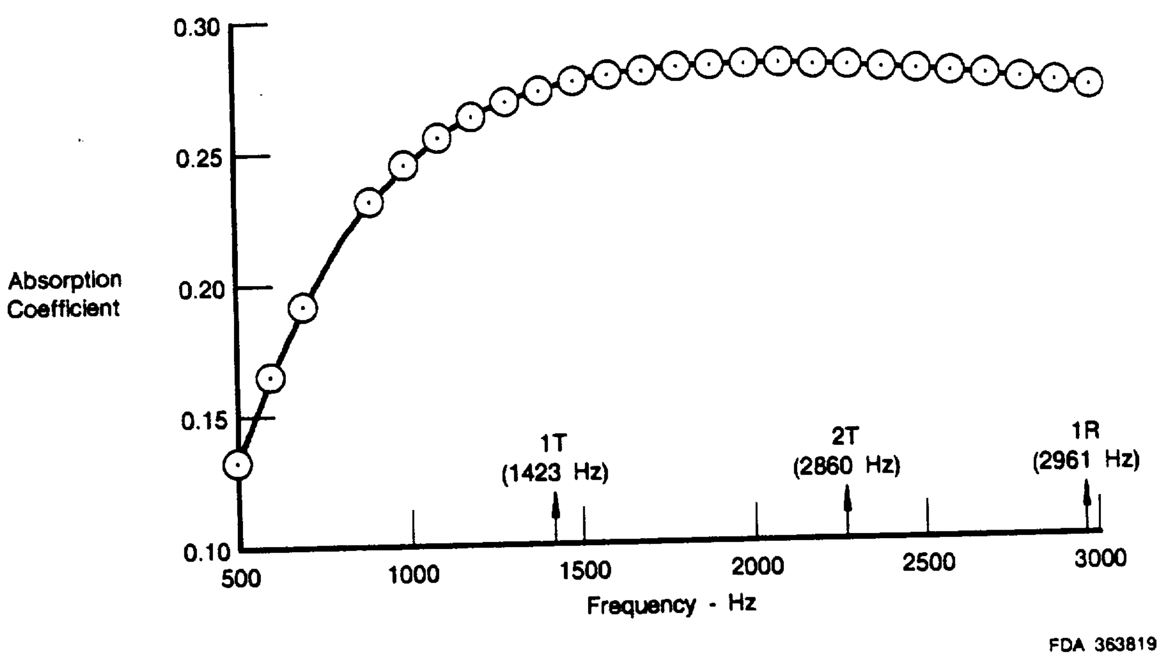

Figure 4.1.1.4-2. Sound Absorption vs Frequency for STBE Derivative Gas Generator Acoustic Liner

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4.1.1.4.1 Main Injector

The main injector design uses 869 coaxial, tangential entry injection elements arranged in a hexagonal concentric pattern in a 20.35 inch-diameter injector face. This type of injector element has consistently demonstrated efficient, stable combustion in all of the P&W high-pressure combustion programs. The main injector assembly is shown in Figure 4.1.1.4-3.

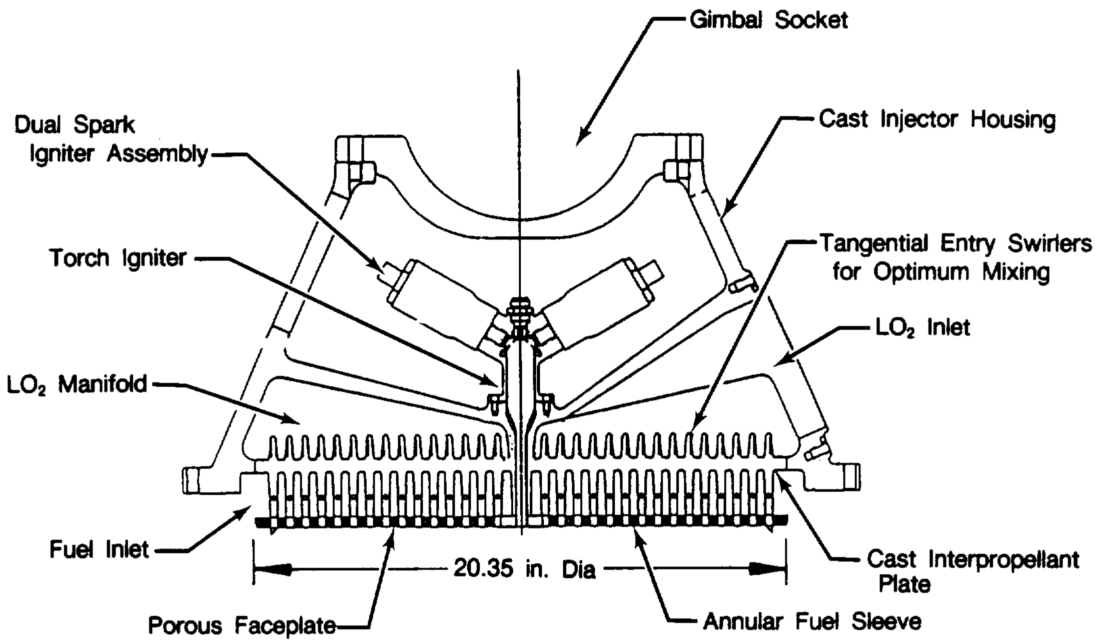

Figure 4.1.1.4-3. STBE Derivative Gas Generator Main Injector Assembly

The oxidizer injection element, shown in Figure 4.1.1.4-4, is a tube which is closed at one end and has a 0.272-inch ID and a 0.020-inch wall thickness. The oxidizer is introduced into the tube through three slots that are oriented on a tangent to the tube ID. The tangential entry produces a hollow cone spray of liquid oxygen which results in extremely fine atomization, and rapid, stable combustion.

Fuel is introduced through an annulus surrounding each LO_2 injection element. The annulus is formed by the fuel sleeve which is cast integral with the injection element and brazed to the porous faceplate. Fuel enters the injector from the combustion chamber coolant interface, and flows radially inward in the injector manifold which is formed by the interpropellant divider plate and the porous faceplate. At each LO_2 injection element, the fuel is directed into the individual fuel annuli by four radial slots in the fuel sleeve. The fuel is then discharged from a 0.02 in.^{2} annulus surrounding each LO_2 injection element. The faceplate is fabricated from a porous material, woven wire product consisting of Haynes 230 cobalt alloy, allowing approximately five percent of the fuel which is introduced into the injector to flow through the injector face to achieve faceplate durability.

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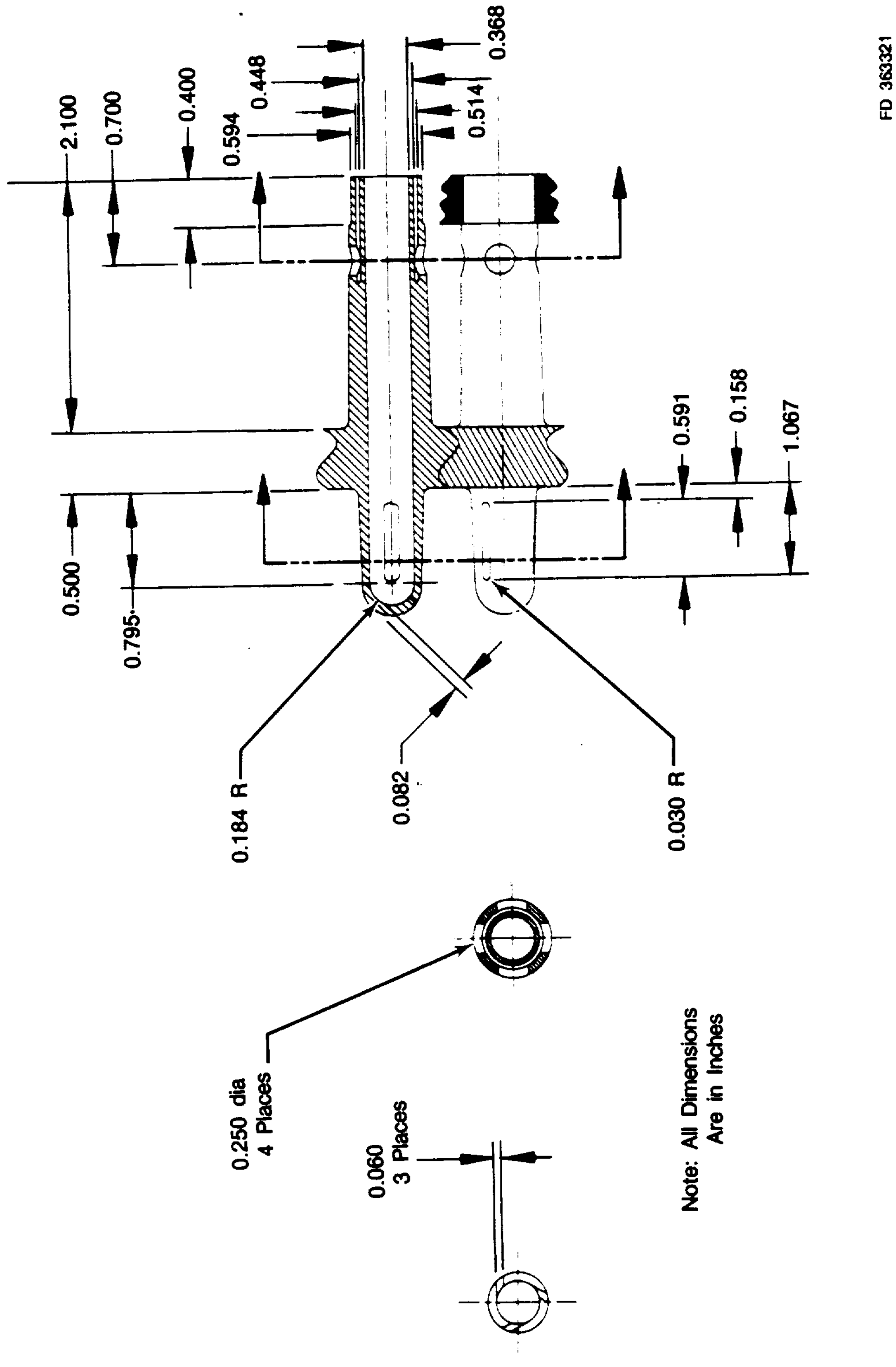

Figure 4.1.1.4-4. STBE Derivative Gas Generator Main Injector Element

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The main injector assembly is fabricated from fine grained cast and HIP Inconel 718 with cast injection elements integral with the propellant divider plate. The injector design provides for a center-mounted torch igniter and also is configured to contain the engine gimbal thrust structure.

4.1.1.4.2 Combustion Chamber

The combustion chamber is regeneratively cooled by fuel from the high-pressure pump discharge. The fuel enters the thermal skin cooling jacket at the regeneratively cooled nozzle manifold chamber interface. The coolant then flows forward, counter to the gas path flow, to the throat. The fuel cools the chamber wall, exits at the injector interface internal manifold, and enters the injector. This flow configuration provides the coolest fuel at the throat where wall heat flux is highest. The combustion chamber is shown in Figure 4.1.1.4-5.

The main combustion chamber uses similar construction technologies as the SSME Main Combustion Chamber in the area of the regeneratively cooled liner. However, the construction differs in the structural jacket design. The regeneratively cooled liner will be forged from NASA-Z copper alloy. The cooling passages are machined from the copper alloy liner and an electrodeposited nickel close-out is applied which forms the outer jacket of the liner. At this point the structural jacket of aluminum is installed around the liner by a bi-cast method. This is accomplished by positioning a sand mold around the liner, then the structural jacket is cast in place with an aluminum casting alloy.

The structural aspect of the bi-cast chamber design is very similar to the conventional welded nickel design. The layer of copper and nickel is used to close out the passages and hold the coolant pressure, and the structural jacket is used to contain the hoop loads due to the combustion chamber pressure. The axial load from the nozzle, i.e., thrust, is also transferred through the jacket by longitudinal webs in the bi-cast aluminum design. A close fit between the copper liner and structural jacket is obtained to ensure that the hoop loads are transmitted to the jacket and do not cause overstressing of the liner.

An acoustic cavity is positioned adjacent to the injector face to provide combustion stability. The acoustic cavity is located behind the copper alloy liner. The cavity is connected to the combustion chamber cavity through a specified number and size of holes through the liner between the coolant passages. A liner is placed in the acoustic cavity to which a minimal amount of coolant flow is tapped off the chamber coolant exit, and used to cool the backside of the acoustic cavity. This coolant is then dumped into the cavity to provide a purged outflow, preventing hot gas ingestion into the acoustic cavity.

The STBE derivative gas generator thrust chamber is a derivative of the STME chamber. The derivative chamber has identical values of manifold location and size, divergent nozzle exit diameter, chamber diameter and injector-to-nozzle exit length as the STME design. The chamber features a machined passage thermal-skin NASA-Z liner/nickel closeout assembly surrounded by a structural jacket. The coolant enters the common inlet manifold and flows counterflow toward the injector, where it discharges directly into the injector. The chamber inlet manifold is common with the tubular nozzle which improves the inlet geometry and reduces inlet pressure drop. Since the chamber is cooled with all the chamber fuel flow, the exit manifold can be eliminated which minimizes the coolant exit pressure drop. Due to the higher thrust requirement of the STBE (636K lbf), the throat diameter has been increased from the STME value of 12.87 inches to 14.43 inches. The chamber contraction ratio of 2.0 is less than that for the STME as a result of maintaining a common injector diameter while increasing the throat diameter. The inclusion of the acoustic liner in the chamber increases the difficulty of cooling the liner with this reduced contraction ratio. To cool the liner within the cycle requirements, the

number of passages has been set at 330 with a maximum passage height-to-width aspect ratio of 5.0. The cooling at the throat has been further improved by designing for coolant side curvature enhancement of the heat transfer film coefficient. Figure 4.1.1.4-6 presents the derivative thrust chamber contour and passage geometry summary.

The coolant passage dimensions have been sized to meet the heat transfer and cycle requirements at the 120 percent thrust design point of 750K thrust and the chamber pressure of 2250 psia. Figure 4.1.1.4-7 summarizes the predicted thrust chamber cooling performance at the 120 percent thrust design point. The chamber liner has been designed so that the maximum hot wall temperature is approximately 1530 R. The maximum wall heat flux at this wall temperature is 55.2 Btu/in.^{2}-sec which occurs one-inch forward of the throat. The coolant side curvature enhancement at the high heat flux point is approximately 35 percent. The coolant enters the liner at 236 R and 4934 psia and exits at 430 R and 2589 psia. The passage geometry has also been sized so that the coolant never exceeds a Mach number of 0.5. Highest Mach number in the derivative chamber is 0.2.

4.1.1.4.3 Torch Igniter

A continuous burning torch igniter was chosen for use in both the gas generators and main combustion system because of the simplicity of the design and reliability in tests. The igniter configuration employed evolved from development efforts since 1957 at Pratt & Whitney and is based on experience gained from the successful RL10 and XLR-129 engine programs.

In the gas generator, the torch is mounted in the combustor wall, two inches axially from the injector face, and expels the hot torch combustion gases at a right angle to the flow path from the gas generator injector, thus providing safe, efficient, reliable ignition of the combustion system. In the main combustion chamber, the torch is mounted axially in the center of the injector, directing the torch down along the centerline of the combustion chamber.

The construction of the torch assembly is discussed in Space Transportation Main Engine Configuration Study, P&W FR-19830-1 Volume II, page 93.

4.1.1.4.4 Gas Generator Combustion System

The gas generator employs a fixed-area injector which injects the fuel and liquid oxygen to provide hot gas for the turbopump turbines. This injector design is the result of experience in hot firings using three generations of high-pressure 250K preburner injectors. Approximately 96 percent of the fuel is injected through the concentric annuli around each oxidizer element. The remaining fuel passes through a porous faceplate to provide transpiration cooling and to hold the combustion process off the faceplate. The gas generator assembly is shown in Figure 4.1.1.4-8.

Liquid oxygen is supplied to the injector from the gas generator oxidizer valve to the injector manifold/dome. Oxidizer flow is injected into the combustion chamber through 199 individual swirler elements. Each element has flow entries machined tangentially to the inner diameter. The fuel is injected into the combustion chamber through radial slots in the element fuel sleeve.

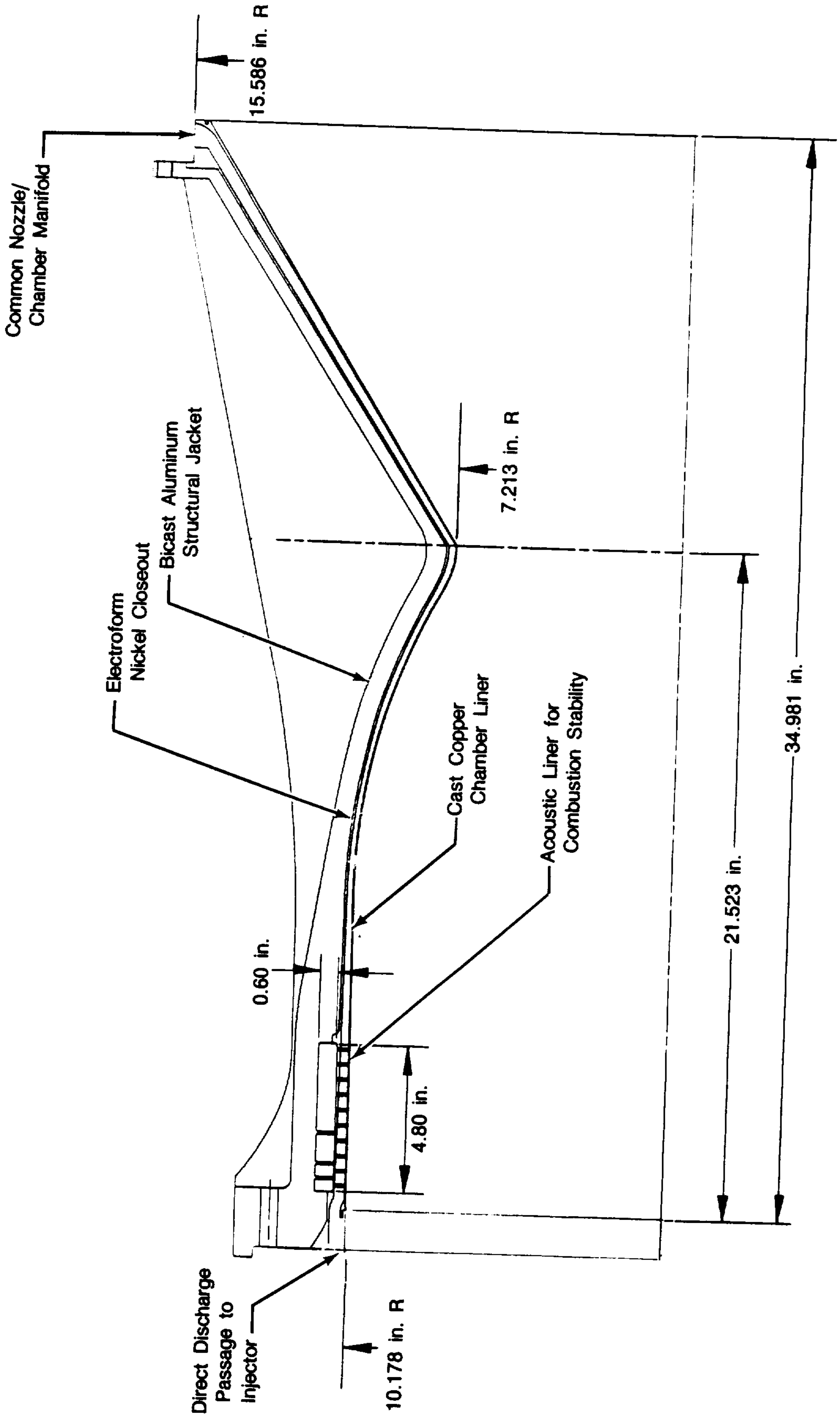

Figure 4.1.1.4-5. STBE Derivative Gas Generator Combustion Chamber

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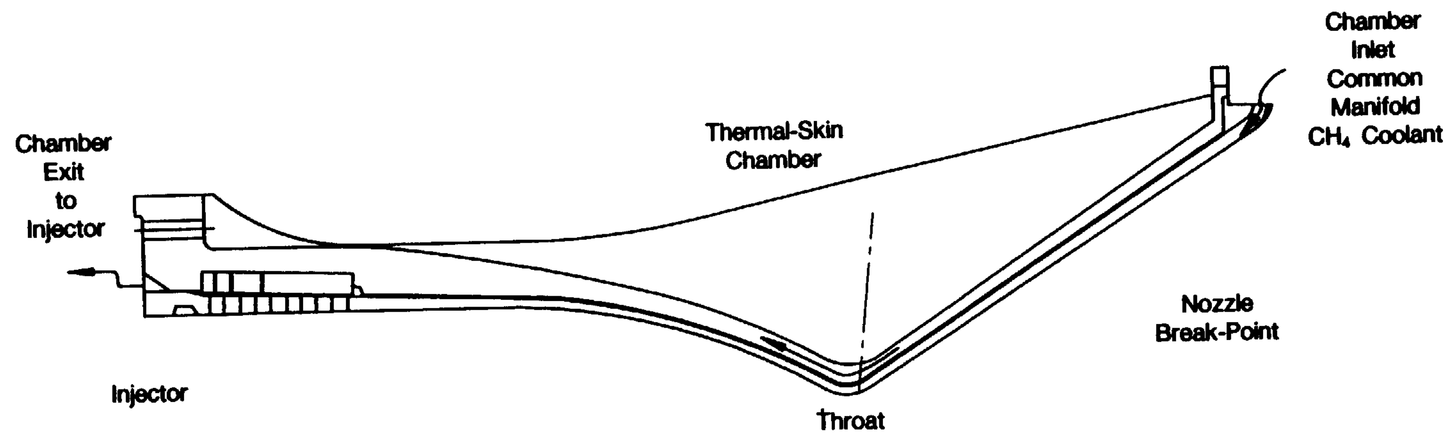

Chamber Contour Data

Chamber Length = 21.523 in.
Divergent Nozzle Length = 13.46 in.
Throat Diameter = 14.43 in.
Injector Diameter = 20.34 in.
Contraction Ratio = 2.0
Divergent Nozzle Area Ratio = 2.16
L^* = 41.0 in.
η_{c^*} (Throat) = 0.98
Number of Passages = 330
Liner Construction - Thermal-Skin
Liner Material - NASA Z

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	Passage Width (in.)	Passage Height (in.)	Land Width (in.)	Wall Thickness (in.)
-21.5	10.18	0.083	0.375	0.111	0.035
-18.1	10.18	0.083	0.375	0.111	0.035
-16.1	11.18	0.083	0.335	0.111	0.035
-12.1	11.18	0.120	0.205	0.074	0.035
-9.1	10.06	0.120	0.186	0.071	0.035
-5.0	9.17	0.102	0.195	0.090	0.035
-1.0	7.39	0.085	0.173	0.054	0.035
-0.0	7.21	0.085	0.168	0.052	0.035
2.4	8.70	0.085	0.340	0.081	0.047
5.4	10.58	0.140	0.324	0.073	0.061
10.4	15.71	0.140	0.467	0.121	0.086
13.5	15.59	0.140	0.480	0.157	0.100

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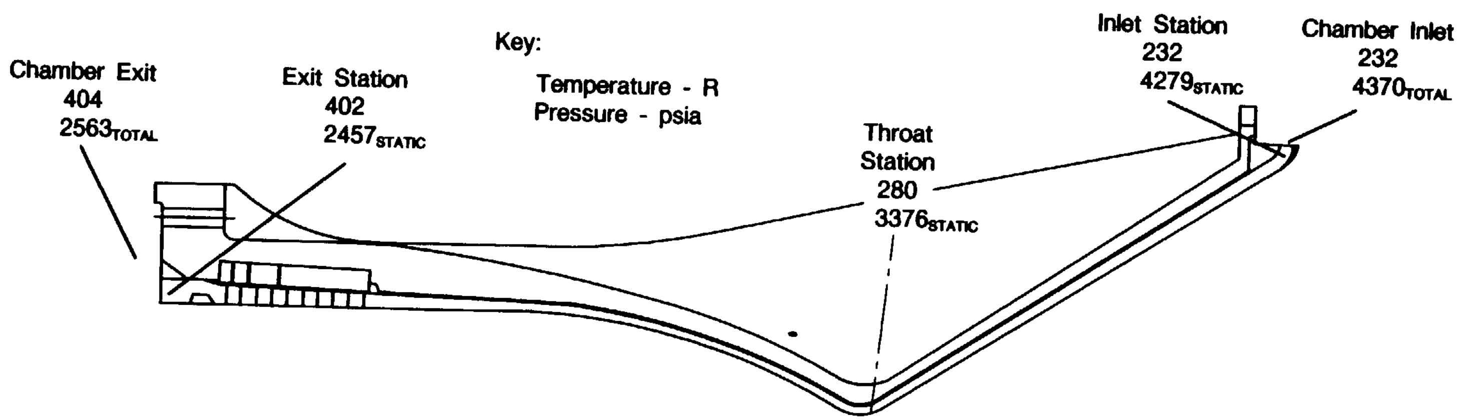
Figure 4.1.1.4-6. STBE Derivative Gas Generator Chamber Cooling Passage Geometry and Contour Data



-21.5	-18.1	-16.1	-12.1	-9.1	-5.0	-1.0	0.0	2.4	5.4	10.4	13.5
932	1471	1506	1496	1435	1411	1502	1502	1289	1509	1446	1377
16.4	30.3	32.5	34.2	35.6	41.1	54.6	55.2	23.3	17.7	12.9	11.2

Coolant Performance

Thrust = 120%
M_{cool} = 442.2 lbm/ sec

Chamber Heat Transfer Performance

Thrust - lbf	637K
Chamber Pressure - psia	2250
Coolant Flow - lbm/sec	442.2
Inlet Temperature - R	232.0
Exit Temperature - R	407.0
Coolant Heat Pickup - Btu/sec	67653.0
Inlet Pressure - psia	4370.0
Exit Presure - psia	2563.0
Pressure Drop - psid	1807.0

Hot Wall Temperature & Heat Flux

Key:

Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2} - sec

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Figure 4.1.1.4-7. STBE Derivative Gas Generator Chamber Heat Transfer Performance Summary

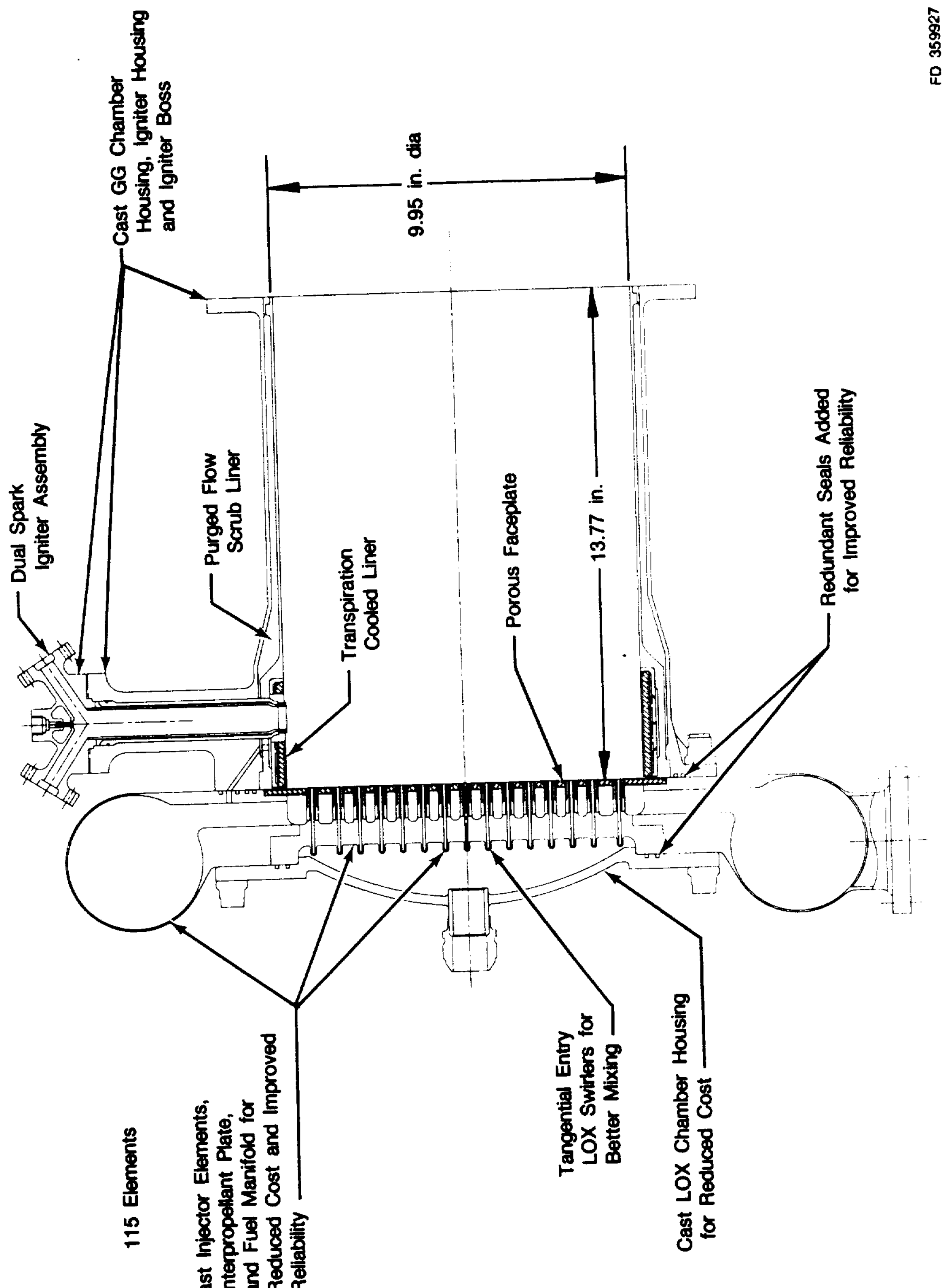


Figure 4.1.1.4-8. STBE Derivative Gas Generator Assembly

The gas generator injector is fabricated from a cast Haynes 230 divider plate with integral injection elements. The oxidizer manifold cavity is formed by a bolted-on dome-shaped end plate. The fuel manifold is formed by a torus welded to the cast divider plate. The porous faceplate is brazed to each injector element fuel annulus sleeve, thereby providing structural support to the plate. The faceplate is made from a porous, woven wire product consisting of Haynes 230 cobalt alloy. This material provides good oxidation resistance and high temperature strength to resist the erosion effects if hot gas scrubbing does occur. The faceplate provides a high enough pressure differential to cause the fuel to uniformly distribute for concentric injection into the sleeve around the oxidizer element, yet passes enough fuel to transpiration cool the material and float the combustion gas away from its surface.

The combustion chamber consists of two basic assemblies, the scrub liner and the structural case. The scrub liner forms the hot gas flowpath and protects the structural case from the hot gas. The scrub liner consists of a porous and non-porous liner. Both are made from Haynes 230 cobalt base material needed for its oxidation resistance and high-temperature capabilities. The front three inches consists of a porous liner that is transpiration cooled by a portion of the fuel flow tapped from the gas generator injector. This front zone is the region of highest energy release, and in addition to providing thermal protection, the porous liner also serves as an effective acoustic damper to prevent combustion instability. The other (non-porous) liner is a cylindrical duct which forms the combustion chamber.

4.1.1.5 Nozzle

4.1.1.5.1 Regeneratively Cooled Nozzle

The regeneratively cooled nozzle, shown in Figure 4.1.1.5-1, is constructed from 990 SPIF (Super Plastic Inflation Formed) tubes of AISI 347 stainless steel, surrounded by a structure shell of closed cell elastomeric foam with a filament wound composite overwrap. This shell is also designed to carry all hoop loads.

The SPIF nozzle is welded to the inlet and exit manifolds which are both made of AISI 347 SST. The closed cell polyurethane foam on the exterior of the nozzle, would adhere to the nozzle surface and act as a compliant layer between the nozzle and the composite structural shell due to the coefficient of expansion difference between the nozzle and shell. At cryogenic operation the foam would become rigid, thereby transferring the nozzle hoop load into the structural shell. The nozzle coolant inlet manifold supplies coolant to the nozzle and the combustion chamber, making the nozzle parallel coolant flow and the combustion chamber coolant counterflow.

The regeneratively cooled nozzle is entirely common with that of the STME. Figure 4.1.1.5-2 summarizes the nozzle geometry. The nozzle is constructed of 990 super plastic inflation formed AISI 347 stainless tubes. The nozzle is 56-inches long and extends from an expansion area ratio of 2.16:1 to an exit area ratio of 27.9:1. The number of passages and the passage diameters have been sized so that the operating stresses of the wall never exceed the 0.2 percent yield stress. An alternate nozzle design could be constructed of 990 Haynes 230 tubes.

The coolant enters the nozzle at an area ratio of 2.16, flows parallel to the gas path flow and exits at an area ratio of 27.8. Figure 4.1.1.5-3 presents the predicted heat transfer performance of the nozzle. the nozzle is cooled with 146 lbm/sec of fuel that enters at 234 R and 4024 psia and exits at 563 R and 3502 psia. The maximum hot wall temperature and heat flux are 1455 R and 10.9 Btu/in.^{2}-sec, respectively.

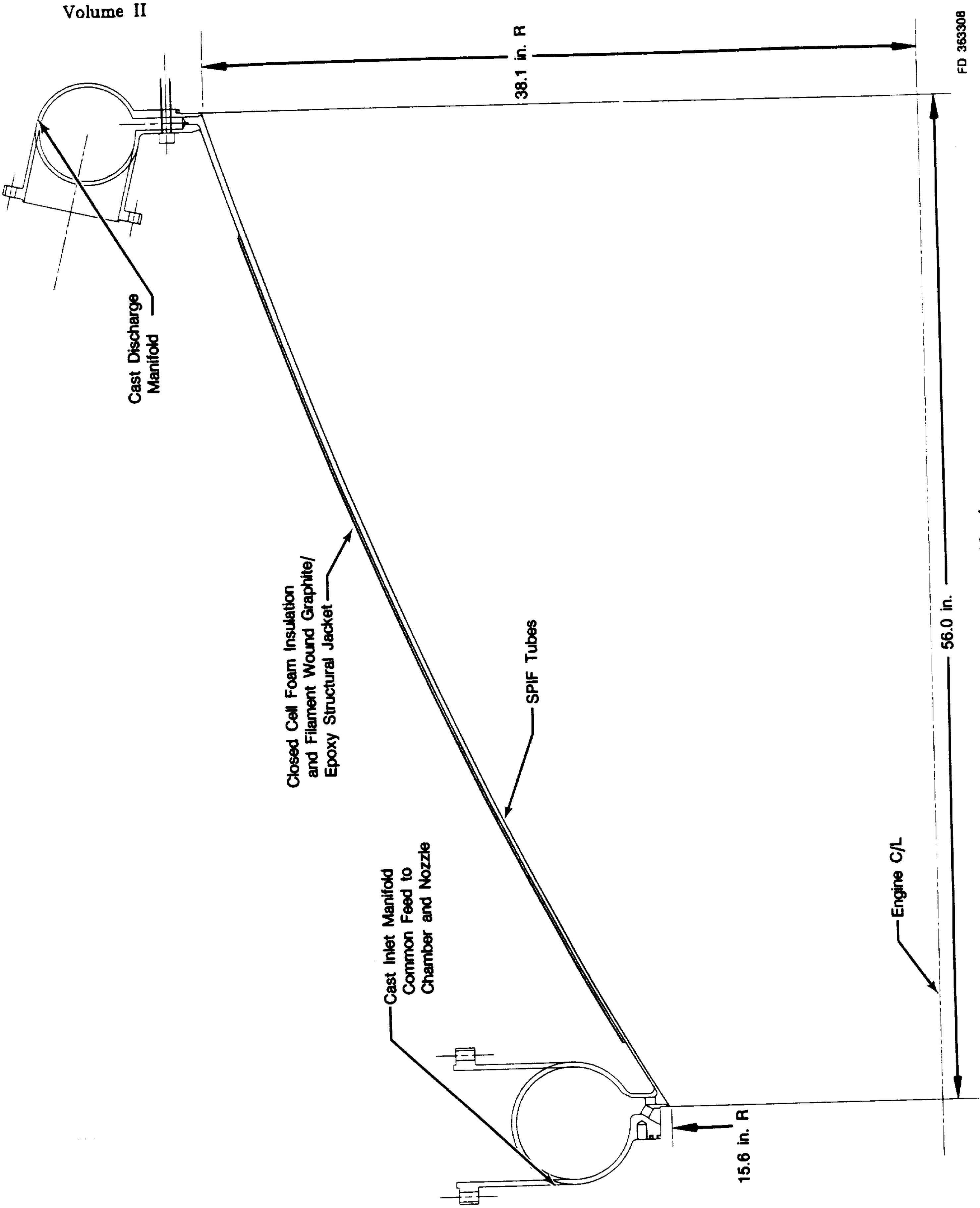
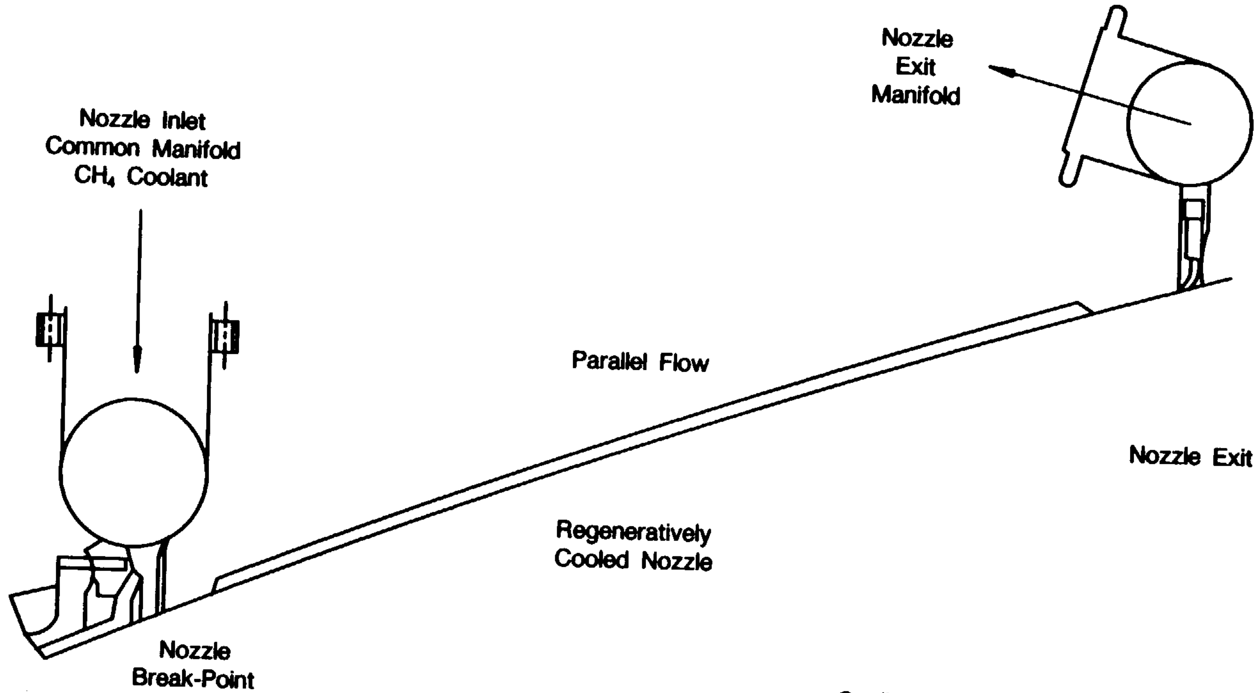


Figure 4.1.1.5-1. STBE Derivative Gas Generator Regeneratively Cooled Nozzle



Nozzle Contour Data

Nozzle Length - 56 in.
Inlet Expansion Ratio - 4.65
Exit Expansion Ratio - 27.87
Number of Passages - 990
Nozzle Construction - Super Plastic Inflation Formed
Nozzle Material - AISI 347

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)
13.5	15.59	0.097	0.105	0.02
25.9	22.11	0.139	0.152	0.02
40.3	28.36	0.179	0.196	0.02
54.7	33.51	0.211	0.232	0.02
69.5	38.08	0.241	0.266	0.02

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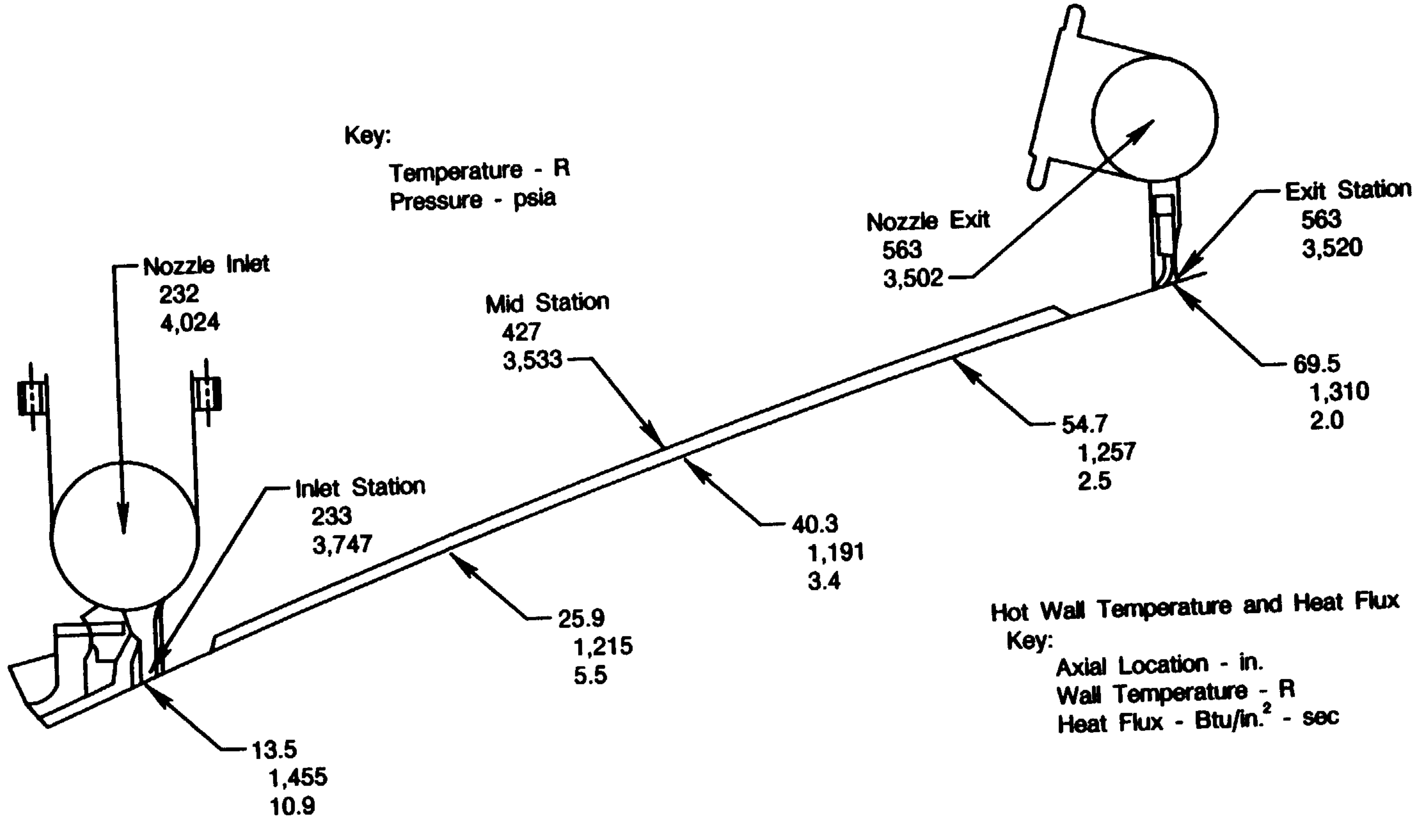
Figure 4.1.1.5-2. STBE Derivative Gas Generator Nozzle Cooling Configuration



Coolant Performance
- Thrust = 120%
- M_{cool} = 146.0 lbm/sec

Nozzle Heat Transfer Performance

Thrust - lbf	645K
Chamber Pressure - psia	2,250
Coolant Flow - lbm/sec	146.0
Inlet Temperature - R	234.0
Exit Temperature - R	563.0
Coolant Heat Pickup - Btu/sec	41,683.0
Inlet Pressure - psia	4,024.0
Exit Pressure - psia	3,502.0
Pressure Drop - psid	522.0

FDA 363806

Figure 4.1.1.5-3. STBE Derivative Gas Generator Nozzle Heat Transfer Performance Summary

4.1.1.6 Gas Generator Engine Control

The STBE control consists of sensors, interconnects, a controller, actuators, propellant valves, ancillary valves, and a health monitor. The functional layout of the STBE control components is shown on Figure 4.1.1.6-1. The controller time sequences the valves for engine control and maintains engine safety by sensing hazards and taking corrective action. A single electromechanical actuator drives both the gas generator fuel and oxidizer valves. The main chamber oxidizer and fuel shutoff valves are helium actuated. The gas generator fuel and oxidizer valves use similar sleeve valves, and the main chamber oxidizer and fuel shutoff valve use similar poppett valves. The health monitor is integrated with the controller but electrically isolated to prevent health monitor faults from propagating into the controller and jeopardizing engine safety.

Engine thrust is regulated by trimming the gas generator oxidizer valve while engine mixture ratio is regulated by trimming the main oxidizer valve. Oxidizer flow shut-off is provided by the gas generator oxidizer valve and the main oxidizer valve while positive fuel flow shut-off is provided by the main fuel shutoff valve.

Requirements used to establish a control and monitoring system concept are shown in Table 4.1.1.6-1.

4.1.1.6.1 Control/Health Monitor Conceptual Architecture

Conceptually the controller/health monitor is comprised of two functions: (1) control and safety monitoring and 2) maintenance monitoring. Control functions are those required to start, maintain normal operating conditions and shutdown the engine. Safety monitoring consists of real time engine evaluation to determine if an emergency shutdown is required. Maintenance monitoring looks at functional and physical characteristics which include many that are not flight critical, but real time definition is necessary to properly schedule maintenance.

The STBE engine uses a simplex, full authority digital electronic engine control with dual channel input/output (I/O). A single channel control with an effector system designed to direct engine shutdown upon loss of controller function meets the fail safe design requirement. Controller reliability requirements are met with dual I/O interfaces which receive inputs from dual sensors with the information being processed by a single microprocessor.

The output interface supports solenoids with dual windings and a dual channel electromechanical actuator interface. One of the two solenoid windings in each device has the capacity for solenoid operation in the event that one winding fails opens. Shorted solenoid switches are accommodated by switching both high and low sides of the solenoid. The electromechanical actuator (EMA) interface is a dual active effector system with single processor control. Under normal conditions, each output interface provides one half the drive signal necessary for actuator control. If one of the EMA interfaces becomes inoperative, the current drivers in the inoperative interface are depowered and the gain in the remaining interface is doubled to provide full control capability. This dual active interface provides smooth transfers from dual channel operation to single channel operation.

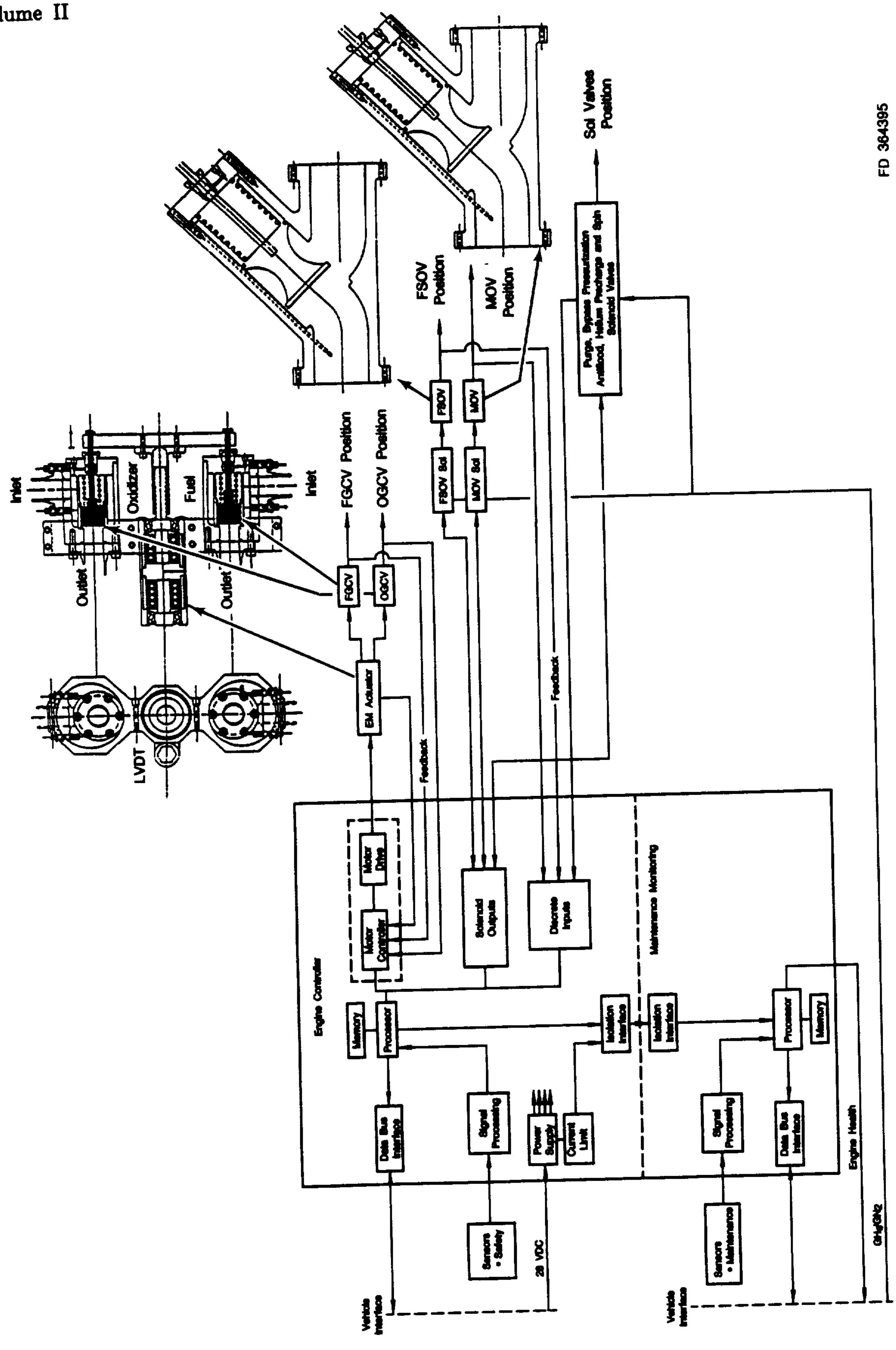


Figure 4.1.1.6-1. ***STBE Derivative Gas Generator Engine Control and Health Monitor System Functional Concept Meets All Requirements With Low-Cost Approach***

Table 4.1.1.6-1. Control System Requirements

Requirement	*Engine Requirement*	*Control System Requirement*
\$300/lb Launch Cost	Low Recurring Costs	Design for Low Costs and Reliability, Provide Prelaunch Checkout
Design Life	5 Hours, 30 Missions	Durability, Maintenance Monitoring
Reliability	Demonstrate 0.99 at 90% Confidence	0.9992
Safety	Fail Safe	Benign Shutdown
Thrust (Vac)	712K ±3%	Ground Trim
Mixture Ratio	6 ±3% at 712K	Ground Trim
Transients	Start to 712K < 5 sec Max Rate of Change of Thrust Shutdown Impulse	Response TBD TBD
Interfaces		
• Tank Pressurization	GCH_4, GO_2	Valves, Logic
• Information	TBD	Data Bus, Baud Rate
• Electrical	N/A	28 vdc
• Ancillary Fluids		
–Ground Operation	Cooldown, Purge	He
–Vehicle Operation	Purge, Actuation	He

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Actuator loop failure detection is provided by current wraparound, feedback failure detection, and open-loop detection. Current wraparound is provided by measuring actuator winding current and comparing the result to the requested value.

Feedback failures occur if the actuator position sensors produce an erroneous result to the controller. Feedback failure detection is provided by detecting out-of-range readings or detecting a difference between the dual sensor readings. Open-loop detection is provided by comparing the requested actuator position to the measured position. The error between the request and feedback is measured over a period of time and compared to a threshold value. If the measured actuator error is above the threshold value, an open-loop failure is declared. In the event that an actuator malfunction cannot be isolated to a given interface, an engine shutdown is effected by the logic.

An initiated built-in-test (IBIT) mode is provided by the controller to detect faults during prestart. In the IBIT mode, the controller sequences solenoid valves and electromechanical actuators throughout their operating range. This feature enhances mission reliability by providing a low cost method for testing the system prior to launch.

The health monitoring system works as an interface between the electronic control, engine sensors, and the vehicle avionics while transmitting real time data to the Vehicle Health Monitoring System (VHMS). Safety monitoring is performed by the electronic control with any performance or anomaly information passed to the maintenance monitoring unit through an

isolation interface. Instrumentation not critical to flight operation is processed by maintenance monitoring electronics. Maintenance monitoring information is transmitted to the vehicle independently of the control.

4.1.1.6.2 Controller Hardware Approach

Highlights of the control/health monitoring system architecture include modular design of the engine control functional requirements. The system level design includes control of discrete inputs and outputs (solenoids and switches), actuator positioning, sensor signal processing and control law processing. This system design is implemented using state of the art hardware which provides a low risk, low cost flexible control.

Current plans are to provide a control design that meets reliability requirements with Class B components. By using these MIL-STD components and proper redundancy management, the reliability requirements can be achieved without the cost penalty of Class S components. With the advent of microelectronics, multiple channel controls are viable options without paying a significant weight penalty. Multiple channel controls will be considered during Phase B as a way to improve life cycle cost.

4.1.1.6.3 Vehicle Interface Definition

Independent vehicle interfaces are supported by both the engine control and health monitor. Independence is necessary to ensure faults in the maintenance data bus from causing a fault in the control data bus. These data buses will be designed to be compatible with the vehicle data bus selection. The only identified differences will be those that address flight criticality. The engine controller interface will be updated to meet different flight safety requirements.

Isolated interfaces between control and maintenance monitor were selected to support the integrated design concept. The key to these interfaces is to incorporate failure containment regions. Failure containment is accomplished through design.

4.1.1.6.4 Actuators/Valves

An extensive trade study was conducted to select valve and actuator types based upon an assessment of cost, reliability, performance and hardware commonality. Low cost was ranked as the primary selection criteria with manufacturability, design simplicity and maintainability all being considered cost drivers. The study considered pneumatic, hydraulic and electromechanical actuators as well as sleeve, poppett, ball, and butterfly valves. From this study, the following configurations were selected.

4.1.1.6.4.1 Ganged Gas Generator Valves/Actuation

The ganged gas generator valve system consists of two valves and an electromechanical actuator. Oxygen and fuel flow to the gas generator are controlled by the Oxidizer Gas Generator Control Valve (OGCV) and Fuel Gas Generator Control Valve (FGCV), respectively. The valves have been ganged together to eliminate potential turbine overtemperature events caused by the OGCV allowing oxidizer flow into the injector following fuel flow shutoff by the FGCV. A linear electromechanical actuator sequences the fuel and oxidizer valves to acheive proper engine start, throttling, and shutdown. Additionally, an oxidizer gas generator bypass valve supplies five percent of oxidizer gas generator flow necessary for starting. This valve is separate from the ganged valve assembly and uses the same concept as the ancillary valves.

4.1.1.6.4.2 Oxidizer Gas Generator Control Valve (OGCV)

Operation

The OGCV is a modulating control valve that is located downstream of the oxidizer pump and upstream of the gas generator injector. Its function is to accurately control oxidizer flow into the gas generator and thereby control the thrust level of the engine. The valve schedules shown in Figures 4.1.1.6-2 and -3 indicate that the valve must accurately meter oxidizer flow for engine start, for engine transition to a second thrust level, and for engine shutdown, and therefore requires a high turndown ratio, or capability to meter accurately over a large range in flow. Evaluation of a valve type to meet these requirements at the lowest cost resulted in the selection of a right angle inlet to outlet translating sleeve type valve for this application, as shown in Figure 4.1.1.6-1. By contouring the sleeve metering ports, the valve area versus stroke relationship may be customized to meet the 2.5 percent accuracy requirement at all engine conditions. To meet the failsafe safety requirements for benign engine shutdown and to minimize required actuator force, the OGCV is pressure balanced and spring loaded in the closed direction. Thus, upon loss of actuator input force, for whatever reason, the OGCV slews to the closed position at a rate controlled by the valve force balance and the flow rate of oxidizer into the pressure balance cavity of the valve assembly.

Fabrication

The sleeve type OGCV design can be fabricated from standard bar stock shapes, allowing the use of simple manufacturing processes and ease of fabrication over a wide supplier base. Also, all parts/assemblies can be made identical to the FGCV with the exception of the sleeve, allowing low cost manufacturing due to increased lot sizes.

To reduce maintenance and improve reliability, ceramic materials are being investigated for the valve and sleeve elements. The material, Zirconia Toughened Alumina (ZTA), has been fabricated into a sleeve and valve configuration by a valve supplier. This valve eliminates the potential risk associated with metal to metal sliding surfaces in LO_2 and initial testing has shown that ZTA erosion and wear characteristics are ten times better than conventional 440 steel. Further investigation, including thermal shock testing, must be completed to determine this material's applicability.

4.1.1.6.4.3 Fuel Gas Generator Control Valve (FGCV)

The FGCV is an on/off valve that is located downstream of the nozzle fuel coolant exit and upstream of the gas generator injector. Its function is to control the flow of gaseous fuel into the gas generator and thereby control the gas generator oxidizer/fuel mixture ratio. To meet the engine start and throttling requirements the valve requires only one full open and one full closed position. Evaluation of a valve type to meet the requirements and provide maximum commonality with the OGCV has resulted in selection of a sleeve type valve identical to the OGCV with the exception of the sleeve which is ported for much higher area versus stroke gain. Since flow area is maximized when the sleeve ports are completely uncovered, the valve element may continue to translate without increasing the actual flow area of the FGCV. Thus, the ganged valve assembly may be postitioned variably to control OGCV position, which controls thrust, without impacting FGCV area. The FGCV is pressure balanced closed and spring loaded closed in a manner identical to that of the OGCV.

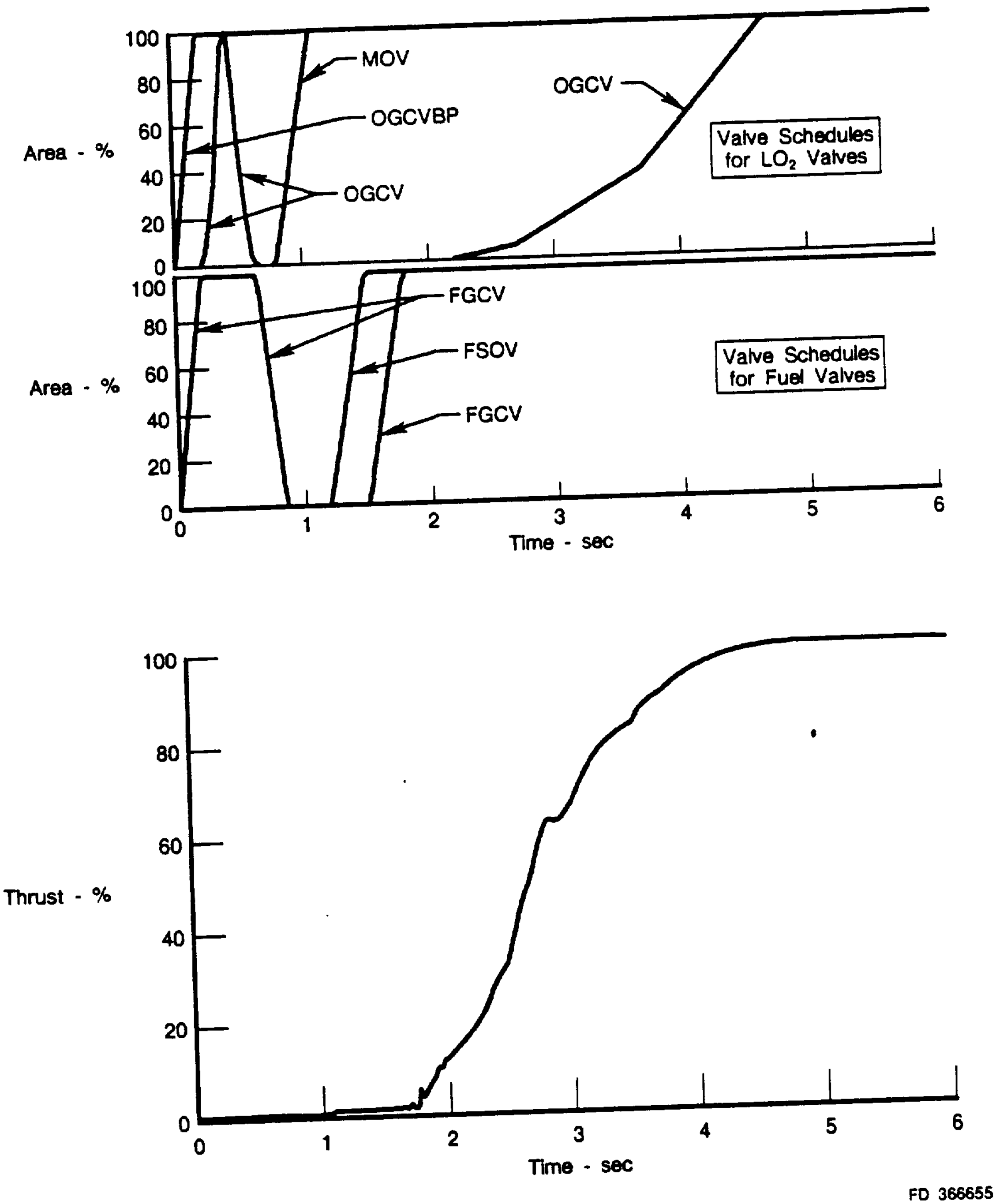

Figure 4.1.1.6-2. Valve Sequence and Thrust Buildup for Engine Start

Fabrication

The FGCV will be fabricated identically to the OGCV and will reduce production cost by allowing larger lot size purchases of the identical FGCV and OGCV parts.

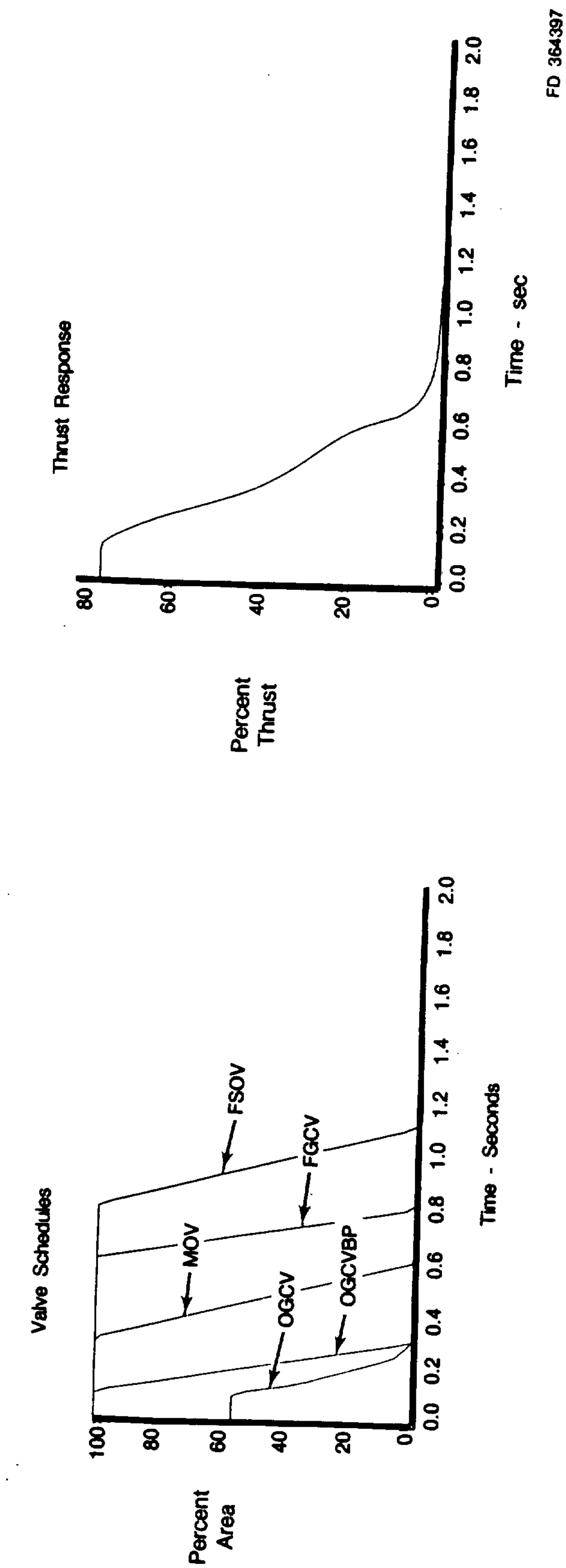

Figure 4.1.1.6-3. Valve Schedule and Thrust Transient for Engine Shutdown

4.1.1.6.4.4 Ganged Valve Actuation

The gas generator valves are ganged for actuation with one actuator to eliminate potential turbine overtemperature events caused by the OGCV remaining open after the FGCV has closed. To satisfy the OGCV variable duty cycle this actuator must provide accurate position scheduling, while also providing a simple preflight checkout procedure. To meet the duty cycle requirements for both oxidizer and fuel flow, the ganged gas generator valves have been sequenced to result in actual area versus stroke characterisitcs as shown in Figure 4.1.1.6-4. This sequencing is permitted by the flexibility of the sleeve contouring and results in flow control as requested in the duty cycle. To provide a benign engine shutdown for the failsafe safety feature, the actuator must fail-passive such that the gas generator valve loading may backdrive the actuator to close both the OGCV and the FGCV. The lowest life cycle cost type of actuation which meets these requirements is electromechanical actuation. Since hydraulic fluid has been eliminated from the actuator, the operational cost of performing preflight checkouts is reduced and the cost of removal and replacement maintenance actions will also be reduced.

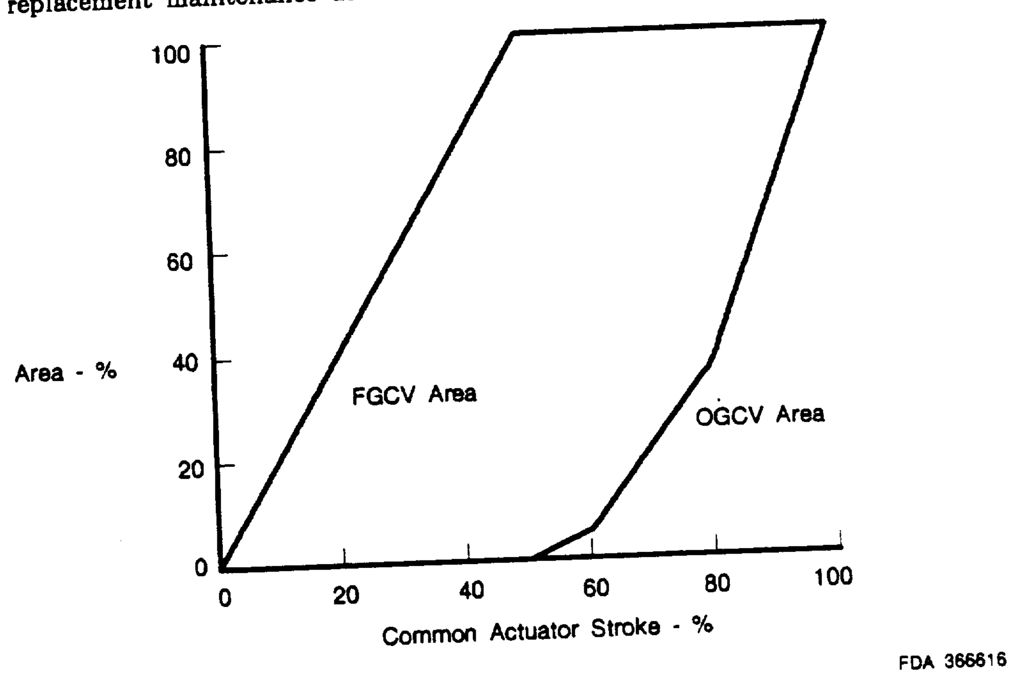

Figure 4.1.1.6-4. Schedule Requirements Feasible With Ganged Valves

4.1.1.6.4.5 Electromechanical Actuator Operation

The electromechanical actuation system consists of a dual channel actuator controller and a linear ballscrew actuator. Electrical power is conditioned by a power conditioner to reduce the magnitude of the DC bus electrical transients and to prevent power surges from affecting module operation. The motor controller receives the position command signal from the engine controller along with the position signal from the actuator feedback module. The microprocessor-based controller provides signals to the motor drive circuit, consisting of appropriately configured power semiconductor switches such as metal oxide semiconductor field effect transistors (MOSFETs).

The actuator module consists of dual switched reluctance motors (SRM) directly coupled to a ballscrew device. By directly driving the ballscrew with the electric motors the gear reduction

element associated with electromechanical actuators may be eliminated. The electromechanical actuator linked with the gas generator valves is shown in Figure 4.1.1.6-1.

4.1.1.6.4.6 Main Oxidizer Valve (MOV)

The MOV is an on/off valve that is located downstream of the oxidizer pump and upstream of the thrust chamber. Its function is to control liquid oxidizer flow to the thrust chamber and thereby control the engine oxidizer/fuel mixture ratio. To meet the engine start and throttling requirements, the valve requires only one full open area position and a fully closed position. The valve must provide ±10 percent trimmability at the open position for engine mixture ratio trimming during the engine acceptance testing. A poppett valve has been selected as the lowest cost valve type which will meet all requirements. As shown in Figure 4.1.1.6-1, the poppett lends itself to precision trimming at the 90 percent open position, allowing accurate mixture ratio trimming. Since the valve has only two operating positions, full open and full closed, a translating helium piston actuator has been selected as the lowest cost option meeting all requirements. The actuator position will be controlled through a solenoid valve which is electrically scheduled by the engine controller. Discrete actuator position switches provide valve position feedback to the controller for preflight checkout as well as for in-flight operation.

MOV Option No. 1

To further reduce system cost and improve the reliability by removing components from the system, an optional propellant actuated MOV has been identified. The poppett valve may be pressure balanced and spring loaded such that a difference between the oxidizer pump inlet pressure and the pump outlet pressure serves as the actuation force on the MOV. This configuration restricts the MOV from easily being checked out during the preflight inspections, however, it reduces the potential of an uncommanded valve closure during main stage operation by removing the solenoid actuator and replacing it with a force balanced poppett assembly. Thus, the MOV will not close until the oxidizer pump pressure delta falls below 300 psid, eliminating the solenoid and actuator failure mode in which the pump is overpressurized as a result of MOV closure at main stage operation.

MOV Option No. 2

The MOV may also be electromechanically actuated to provide active mixture ratio trim during engine operation. Using the pressure balance technique, the valve loads may be reduced such that the electromechanical actuator used for the ganged valve assembly may also be used for the MOV.

4.1.1.6.4.7 Fuel Shutoff Valve (FSOV)

The FSOV is an on/off valve that is located downstream of the fuel pump and upstream of the nozzle and chamber coolant jackets. Its function is to control the total fuel flow into the engine cycle. To provide maximum cost benefit, a poppett type valve identical to the MOV has been selected. While pressure drop and weight could be improved using a ball valve design in this location, these factors have been traded for the simpler, lower cost poppett which also provides commonality with the MOV and the cost benefits which accompany commonality in development, production and logistics. The actuator is identical to that of the MOV providing additional system commonality. The actuator position will be controlled through a solenoid valve which is electrically scheduled by the engine controller. Discrete actuator position switches provide valve position feedback to the controller for preflight checkout as well as for in-flight operation.

4.1.1.6.4.8 Ancillary Valves

To provide propellant purging upon engine shutdown, tank pressurization during engine operation, pump interstage dam pressurization, and oxidizer gas generator valve bypass, solenoid actuated ancillary valves will be used. In each case the valves are low cost poppett type valves which require only short stroke actuation. For the propellant purge valves, a check valve is located between the poppett and the propellant line to help insure that the propellant is isolated from the helium system. These valves will incorporate commonality when possible, however, sizing and failsafe requirements for each valve must be defined before the degree of commonality can be established. Each ancillary valve will provide valve position feedback to the controller using dual valve open and valve close switches.

4.1.1.6.4.9 Operation

Valve/solenoid/ignition sequencing during prestart, start, mainstage, shutdown and post shutdown (in-flight) are shown in Figure 4.1.1.6-5.

4.1.1.6.4.10 Prelaunch Checkout

All valves are stroked from full closed to full open to full closed. Valve slew times provide verification that the valves are operational.

4.1.1.6.5 Pumps Cooldown

The turbopumps are cooled to cryogenic temperatures by liquid hydrogen and liquid oxygen supplied through the vehicle inlet lines. Other than activating purge flows no control valve sequencing is required by the engine.

4.1.1.6.6 Start

The engine start is a timed sequence process using a LO_2 lead for both the gas generator (GG) and main chamber (MC). In the LO_2 lead concept GG and MC fuel is delayed until the injector volumes are filled and liquid oxygen flow is established. This results in a smooth start and eliminates the potential temperature spikes and combustion instability associated with two phase LOX injector flow.

Helium is introduced to the GG via the GG fuel injector simultaneously purging any oxygen from the fuel injector and providing helium spin up assist to improve start repeatability and help in achieving the 5 second start requirement. Figure 4.1.1.6-2 shows the valve scheduling and thrust building characteristic during the start. Thrust buildup rates can be tailored to meet the start requirement by modifying the GG valve start schedule. A bypass valve (OGCVBP) is used to provide LO_2 starting flow prior to opening the GG valves. Fuel rich torches are used for ignition of both the gas generator and main chamber. The use of a fuel rich torch is compatible with safe, fast and reliable ignition when an LO_2 lead start is used.

4.1.1.6.7 Main Stage

Main stage engine operation is an open-loop process. Analysis has shown that an open-loop control concept can be used to meet the ± 3.0% thrust, and mixture ratio requirement, at constant inlet pressure, once the engine is trimmed at the 712K thrust point during the acceptance test. Engine mixture ratio and gas generator mixture ratio are remotely trimmed during engine acceptance testing by trimming the full open position of the MOV and FGCV respectively.

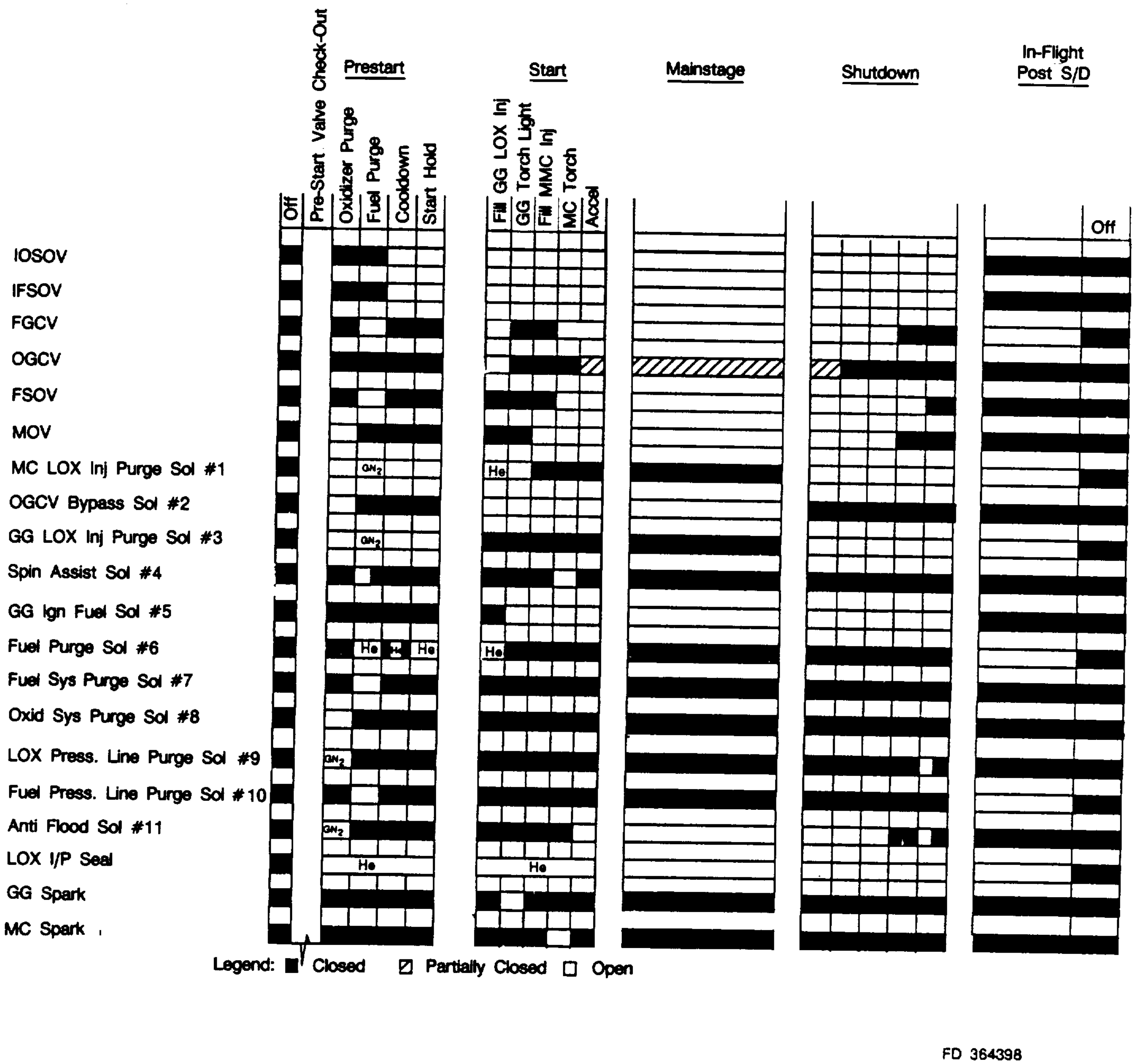

Figure 4.1.1.6-5. Valve Sequencing Accomplished With Timed Logic

4.1.1.6.8 Shutdown

Shutdown is performed by scheduling the propellant valves closed. The OGCV and the OGCVBP are closed first to power down the turbopumps. The MOV and the FGCV are then closed. The FSOV, which shuts off all fuel flow to the engine, is closed last, thus completing the shutdown sequence.

The gas generator and main chamber LO_2 injector purge solenoid valves are opened when the shutdown signal is received from the vehicle. Check valves are included to prevent backflow into the purge lines. When LO_2 injector pressure drops below the checked helium supply pressure the helium purge flow will commence. This flow purges any LO_2 trapped downstream of the OGCV and MOV after they are closed.

Predicted characteristics of an engine shutdown from 712K thrust level are shown in Figure 4.1.1.6-3.

4.1.1.6.9 Post Shutdown

Fuel downstream of the fuel shutoff valve (FSOV) is purged out through the main chamber and fuel gas generator control valve (FGCV). Fuel upstream of the fuel shutoff valve (FSOV) and oxygen upstream of the main oxidizer valve (MOV), oxidizer gas generator control valve (OGCV) and OGCV Bypass is allowed to percolate back to the propellant tanks.

4.1.1.7 Engine Configuration and Integration

4.1.1.7.1 Derivative STBE Gas Generator Engine Assembly

The arrangement of the external configuration of the engine was based on consideration of accessibility for routine component inspections, removals and replacements. Figures 4.1.1.7-1 and -2 show the side and top views of the engine assembly and its major components.

Turbopumps are oriented on a vertical axis and cooldown recirculation valves have been eliminated, leading the way to cooldown by percolation. Engine propellant inlets accommodate engine gimballing through the use of scissor bellows mounted directly to the pump inlets. A toroidal shaped POGO accumulator has been incorporated between the LO_2 pump inlet and the scissors bellows. The engine thrust vectoring gimbal is incorporated into the main injector thrust structure. The gimbal design is based on a ball and socket feature with a central through-pin which restrains torsional movement. A teflon impregnated fiberglass fiber woven fabric between the gimbal ball and main injector socket is used as a friction reduction medium to permit engine gimballing. Gas generator/turbine exhaust is ultimately dumped overboard through the GO_2 heat exchanger and nozzle.

All pneumatic and electrical interfaces are located at the engine interface plane, similar to the SSME.

4.1.1.7.2 Flex Joints

The baseline ALS engine designs use four types of flexible flow ducting joints, bipod stabilized bellows inlet ducts, internally restrained bellows joints, externally restrained bellows joints, and unrestrained compression joints.

The baseline designs which do not use boost pumps result in pump inlets located 33 and 34 inches from the gimbal centerline. Bipod stabilized bellows inlet ducts were selected due to their lower cost and lighter weight when compared to SSME type wraparound articulated ducts. To accommodate the large axial and angular deflections resulting from the 12-degree square pattern gimbaling requirements, the number of bellows convolutions and convolution height were iterated to obtain sufficient flexibility for deflection capability while retaining adequate bellows axial stiffness to prevent squirm due to internal pressure.

The resulting inlet ducts consist of two three-ply Inconel 718 10-inch long bellows with 25 one-inch tall convolutions per duct. These ducts have been designed for a nominal gimbal capability of ± 6 degrees. However, analyses have been conducted to evaluate increases in gimbal capability up to ± 12 degrees. Stabilizing linkages separate the two bellows to prevent buckling of the duct assembly. Excursion limiting stops are included on the stabilizing links to prevent overdeflection of the bellows. Preliminary analysis indicates that at the 12-degree gimbal level, this configuration meets stress criteria but has little margin for bellows squirm. Future analysis is

required to optimize the bellows configuration to minimize the stress levels and to provide additional squirm margin. Vibration analysis is needed to evaluate the potential for flow induced vibration resulting from the vortex shedding phenomena. Some internal bellows damping effect is anticipated due to the three-ply bellows construction. Internal flow guides will be considered, however, their use is complicated by the large axial deflections resulting from the 12-degree gimbaling.

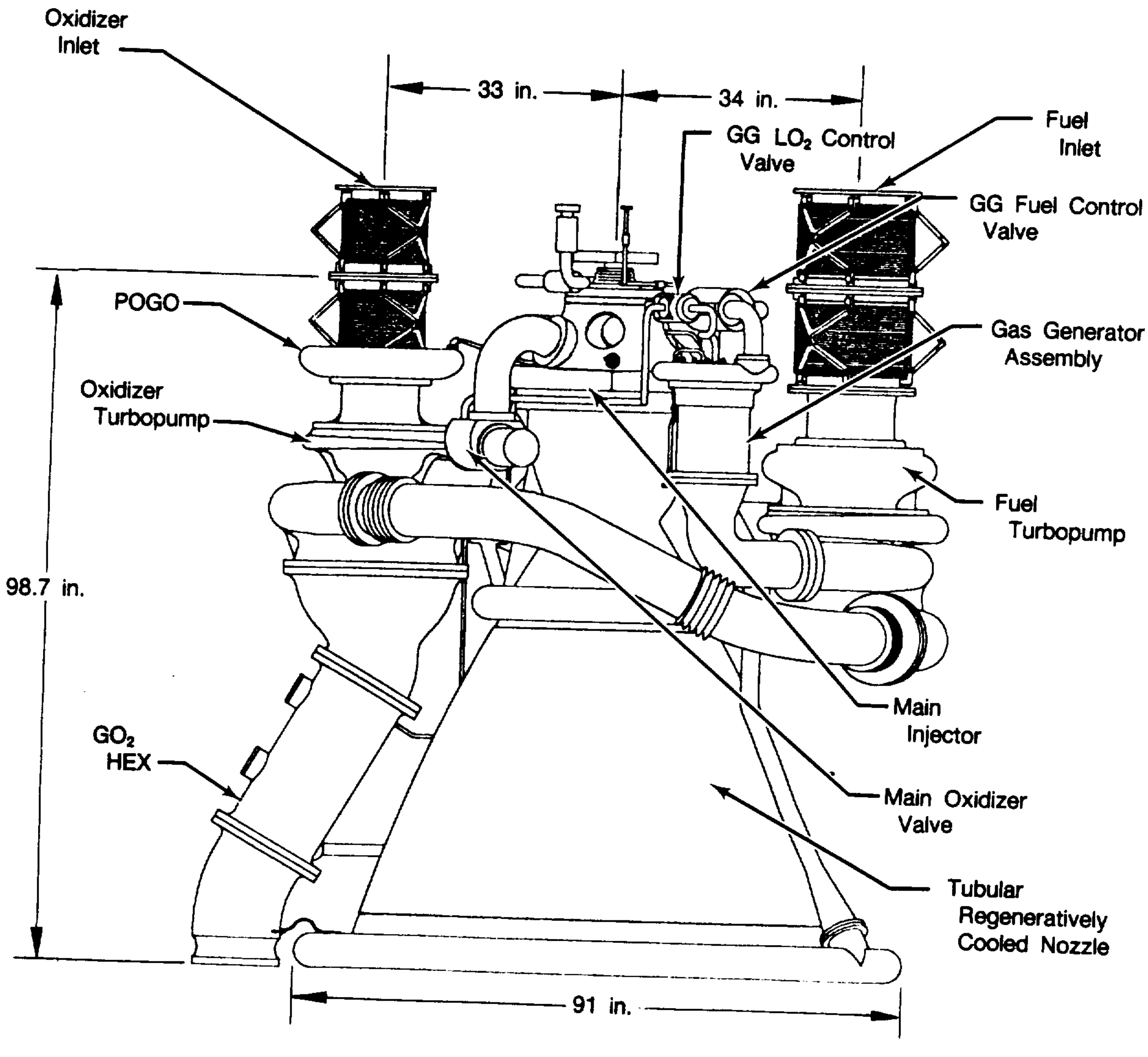

Figure 4.1.1.7-1. STBE Derivative Gas Generator Engine Assembly — Side View

Approximately two degrees of torsional deflection is required on the duct during maximum gimbaling. A low spring-rate bellows torsional spring will likely have to be incorporated in the duct assembly to prevent overstressing of the bellows or pump inlet housings.

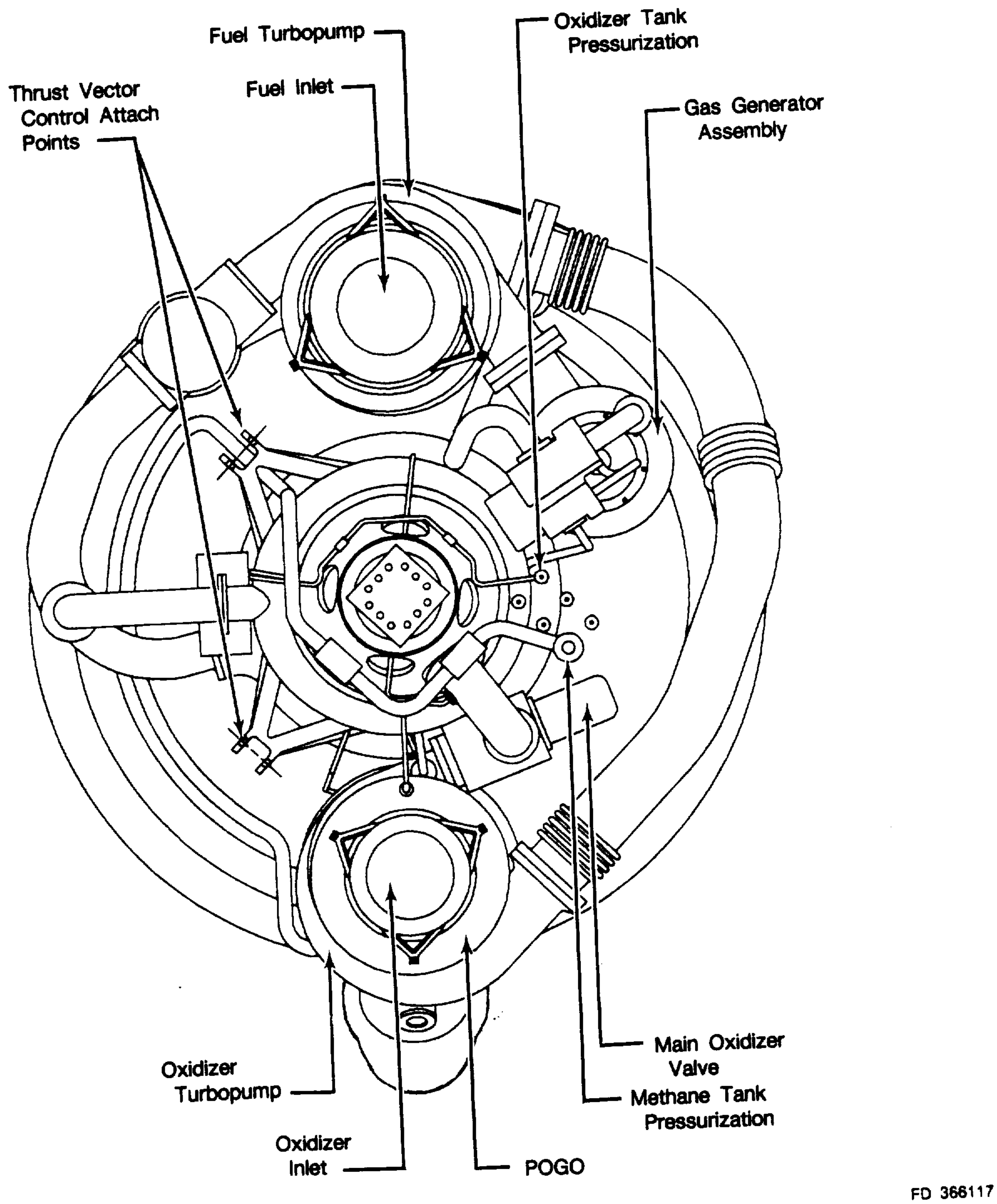

Figure 4.1.1.7-2. STBE Derivative Gas Generator Engine Assembly — Top View

An additional consideration is the large percentage volume change which occurs in the duct during severe gimbaling. If the resulting flow pulse in the LO_2 duct causes significant thrust oscillations, the use of pressure-volume compensating ducts as used on the F-1, or wraparound articulated ducts will have to be evaluated.

In the event that the bipod stabilized ducts prove unsatisfactory for gimbal capability greater than ±6 degrees, after future analysis, wrap around articulated ducts will likely be

chosen for the inlet or intermediate pressure ducts. Three types of gimbal joints were studied for possible inclusion in these ducts: internal ball strut joints, externally pinned joints, and external ball race joints.

The internal ball strut joint, shown in Figure 4.1.1.7-3, contains a ball and socket joint supported by struts in both halves of the joint which guides the joint angulation. A bellows encloses the entire joint assembly. The bellows must carry torsion loads which can cause bellows column buckling when deflected. The main advantage of this configuration is its light weight and small volume. The small envelope size allows it to be easily vacuum jacketed for use in liquid hydrogen ducts. Its simplicity allows it to be the most inexpensive joint while achieving a high degree of reliability. Due to its low torsional load carrying capabilities, its use will likely be limited to hydrogen ducting since the higher density of methane or LO_2 may produce excessive torsional loads on the joint under g-loading. This joint is also used as the baseline for intermediate pressure hot gas flow ducting between turbopump turbines to allow thermal growth in the hot lines.

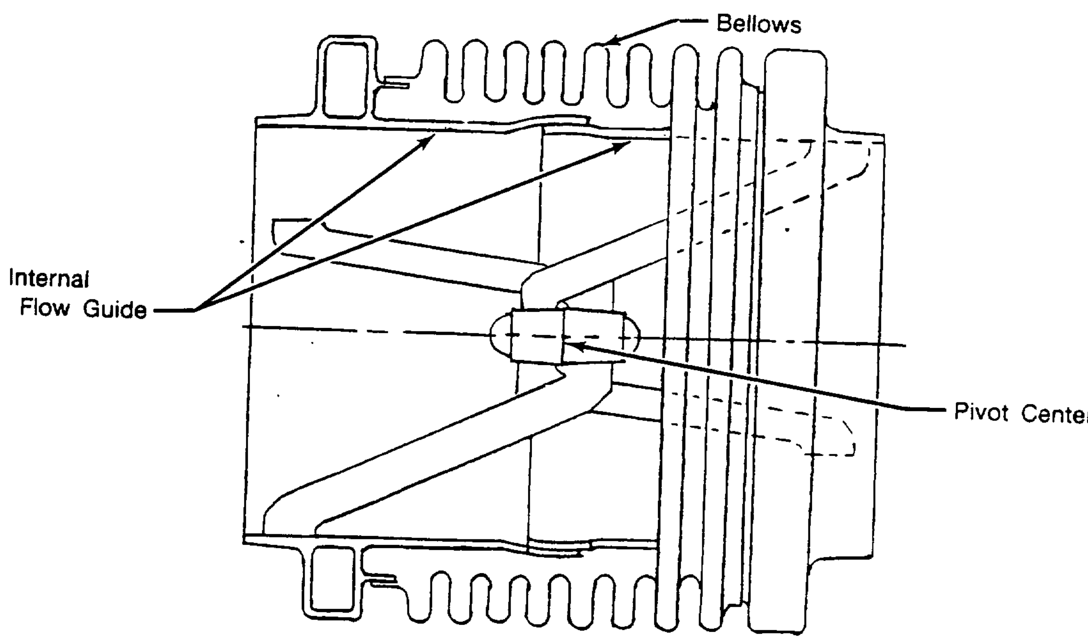

Figure 4.1.1.7-3. Internal Ball Strut Ducting Gimbal

The externally pinned joint, shown in Figure 4.1.1.7-4, uses a universal joint on the outside of the joint which carries all torsional loads. The flow bellows are not subject to torsion loads. The main advantage of this configuration is its low pressure drop due to the lack of obstructions in the flowpath. This configuration has the highest torsional loading capability of the three candidates making it the choice for ducting the higher density fluids. The joint is marginally heavier than the internal ball strut joint and displays similar reliability levels.

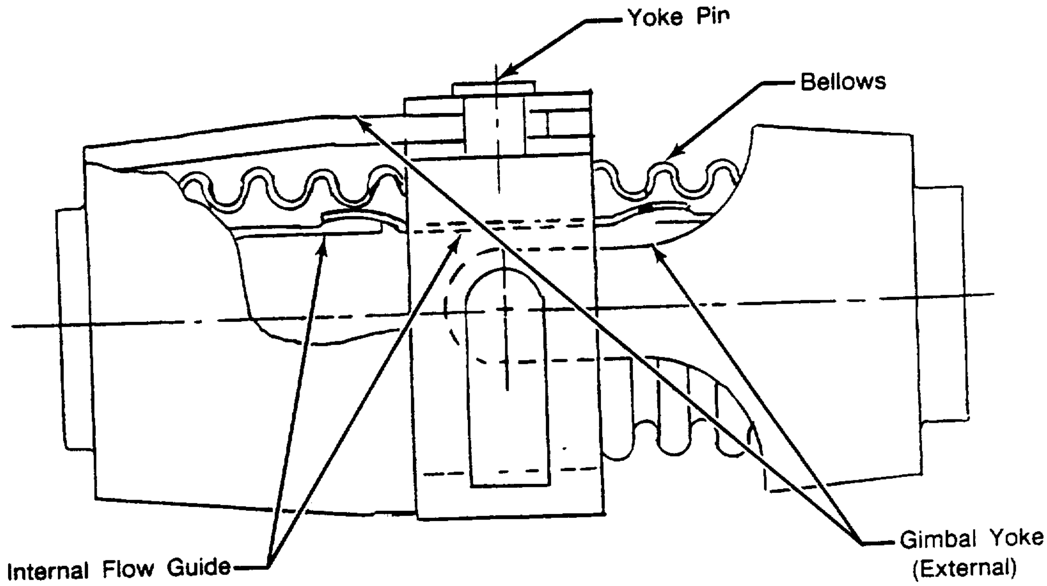

Figure 4.1.1.7-4. Externally Pinned Ducting Gimbal

The external ball race, shown in Figure 4.1.1.7-5, is fastened on the upstream side of the joint to a spherical shell and the downstream side of the joint is fastened in an inner spherical shell. The two shells are separated by ball bearings to reduce friction and are pressure-loaded together to guide the bellows during deflection. This design configuration is the heaviest and provides the lowest angulation levels of the candidate joints and therefore has been eliminated from further consideration. As the bellows carries torsional loads in this design it also has limited torsional capability.

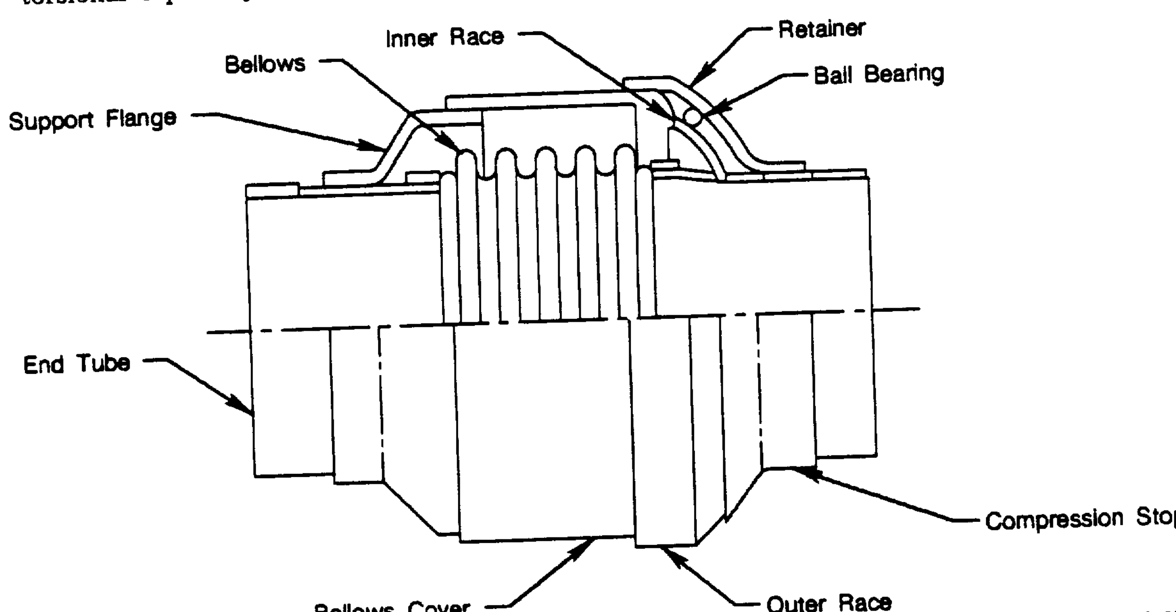

Figure 4.1.1.7-5. External Ball Race Ducting Gimbal

Both of the baseline joints are capable of ± 15 degrees of angulation which should be adequate to allow 12-degree engine gimbaling in wraparound duct configurations. Internal flow liners will maintain acceptable flow characteristics and minimize flow induced bellows vibration.

Unrestrained bellows joints are used in the low pressure turbine exhaust to allow thermal expansion of the ducts. Due to the low pressures, the axial loads transmitted into the mating duct and manifold are low enough to not require a restrained bellows. Care must be taken in designing these ducts to ensure efficient load transfer from the bellows into the surrounding hardware. If the operational deflections of the engine components are large enough, these ducts may be installed in an opposite deflection (loaded) position to allow the duct to move toward a neutral and lower stress position during operation.

4.1.1.7.3 GO_2 Heat Exchanger

The STBE GO_2 heat exchanger, which is common with the STME GO_2 HEX, has been designed to provide gaseous oxygen to the oxygen tank for tank pressurization. The GO_2 heat exchanger uses the gas generator exhaust duct flow as the heat source to vaporize the liquid oxygen as shown in Figure 4.1.1.2-1. The heat exchanger surface is provided by three Haynes 214 stainless steel tubes wrapped in parallel around the gas generator exhaust duct. The gas generator exhaust duct wall is made of beryllium copper with trip-strip roughened walls to enhance the heat transfer. The tubes are packed in powdered copper to structurally isolate the tubes from the duct wall, while providing a good heat transfer medium. This heat exchanger design eliminates the possibility of accidental mixing of the oxygen and gas generator exhaust flow, thereby eliminating a category 1 failure mode.

The GO_2 heat exchanger will require three ⅜-inch diameter tubes 50-feet long, wrapped around the 12-inch duct. The tubes have 0.015-inch thick walls and are separated from one another by 0.055 inch, requiring a total duct length of 1.5 feet. Figure 4.1.1.7-6 diagrammatically presents the GO_2 heat exchanger geometry. The GO_2 heat exchanger has been thermally analyzed for the STBE engine operating point of 100 percent thrust. The oxygen flow rate is predicted to be 5.0 lbm/sec. The heat exchanger has been designed to supply 850 R oxygen to the tank. Figure 4.1.1.7-6 also summarizes the predicted heat exchanger performance.

4.1.1.7.4 Reliability and Maintainability

4.1.1.7.4.1 Reliability

This section provides a complete preliminary Failure Modes and Effects Analysis (FMEA) for the Space Transportation Booster Engine (STBE) Gas Generator Cycle. The section includes the definitions and details used to perform the analysis.

Introduction

Failure Mode and Effects Analysis (FMEA) for the STBE Gas Generator Cycle Engine has been prepared to identify those items that are essential to engine operation. Engine components were analyzed to identify potential failure modes, determine their effects on engine operation, and rank the effects according to Condition Classification. The complete FMEA's are presented in Figure 4.1.1.7-7 and Table 4.1.1.7-1.

STBE GO_2 Heat Exchanger
Thermal and Flow Performance

Gas Generator Exhaust Duct	
Thrust - %	100
Temperature - R	1550
Pressure - psia	289.1
Flow Rate - lbm/sec	185.4
Oxygen	
Flow Rate - lbm/sec	5.0
Inlet Temperature - R	179
Exit Temperature - R	850
Inlet Pressure - psia	3229
Exit Pressure - psia	2889

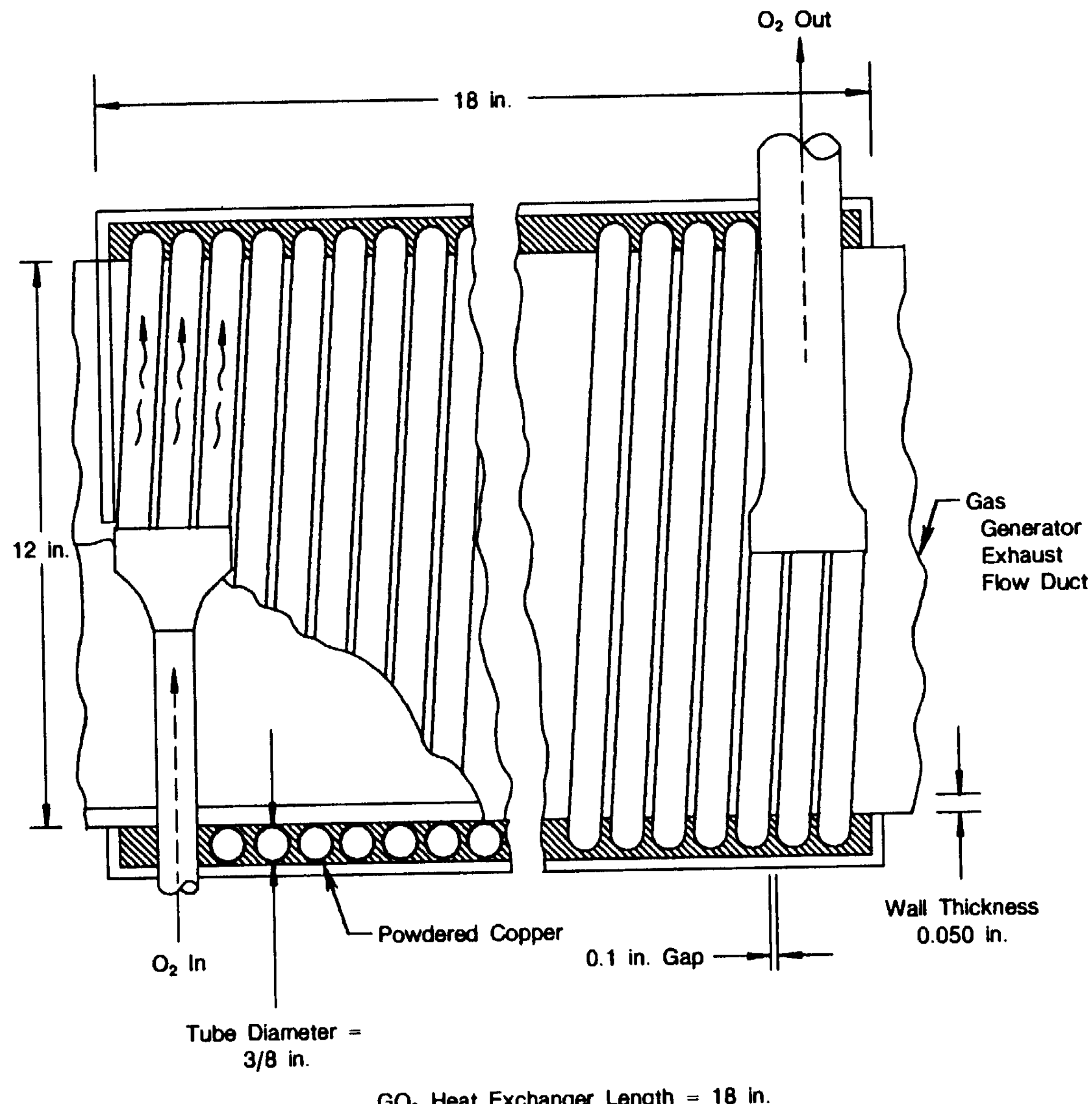

Figure 4.1.1.7-6. STBE Derivative Gas Generator GO_2 HEX Geometry and Performance Data

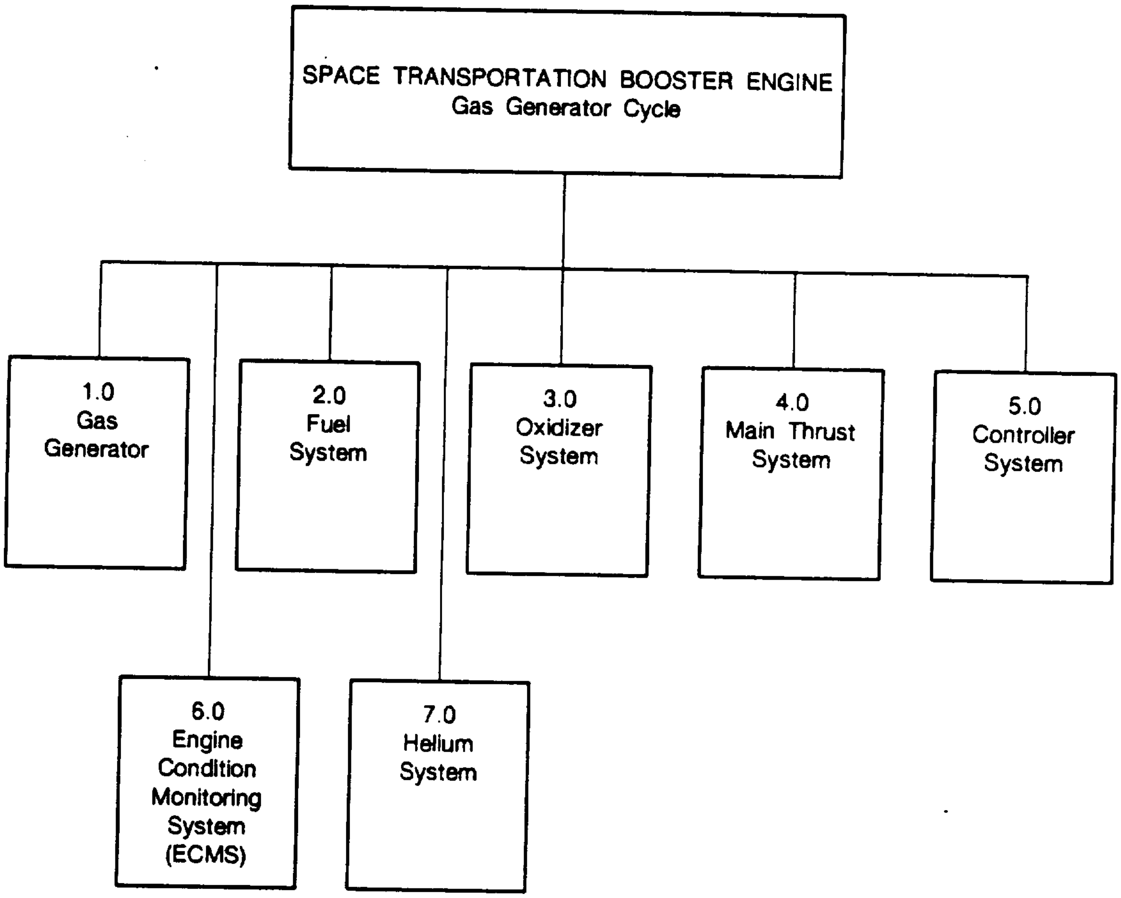

Figure 4.1.1.7-7. Preliminary FMEA — Major Engine Sections (Sheet 1 of 4)

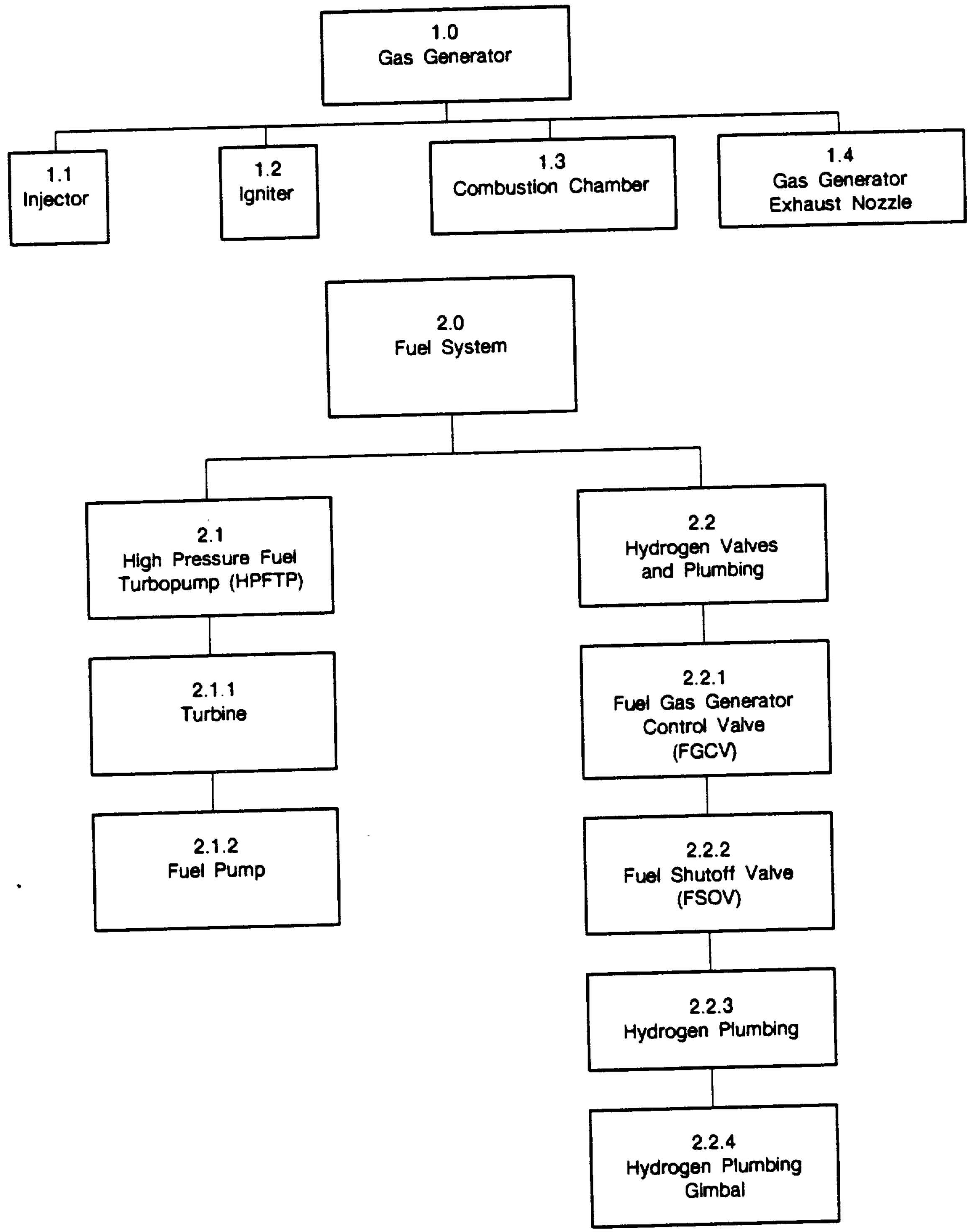

Figure 4.1.1.7-7. Preliminary FMEA — Gas Generator and Fuel Systems (Sheet 2 of 4)

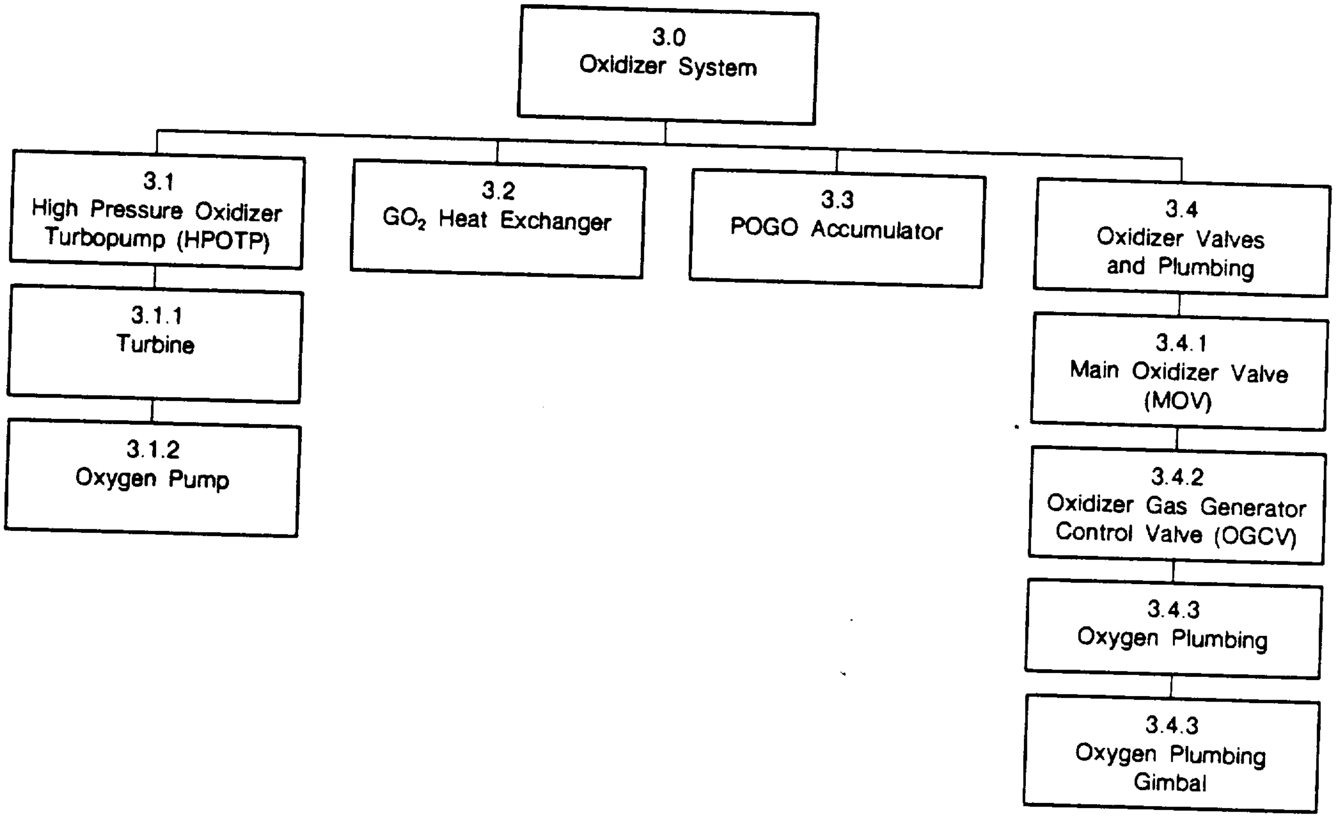

Figure 4.1.1.7-7. Preliminary FMEA — Oxidizer System (Sheet 3 of 4)

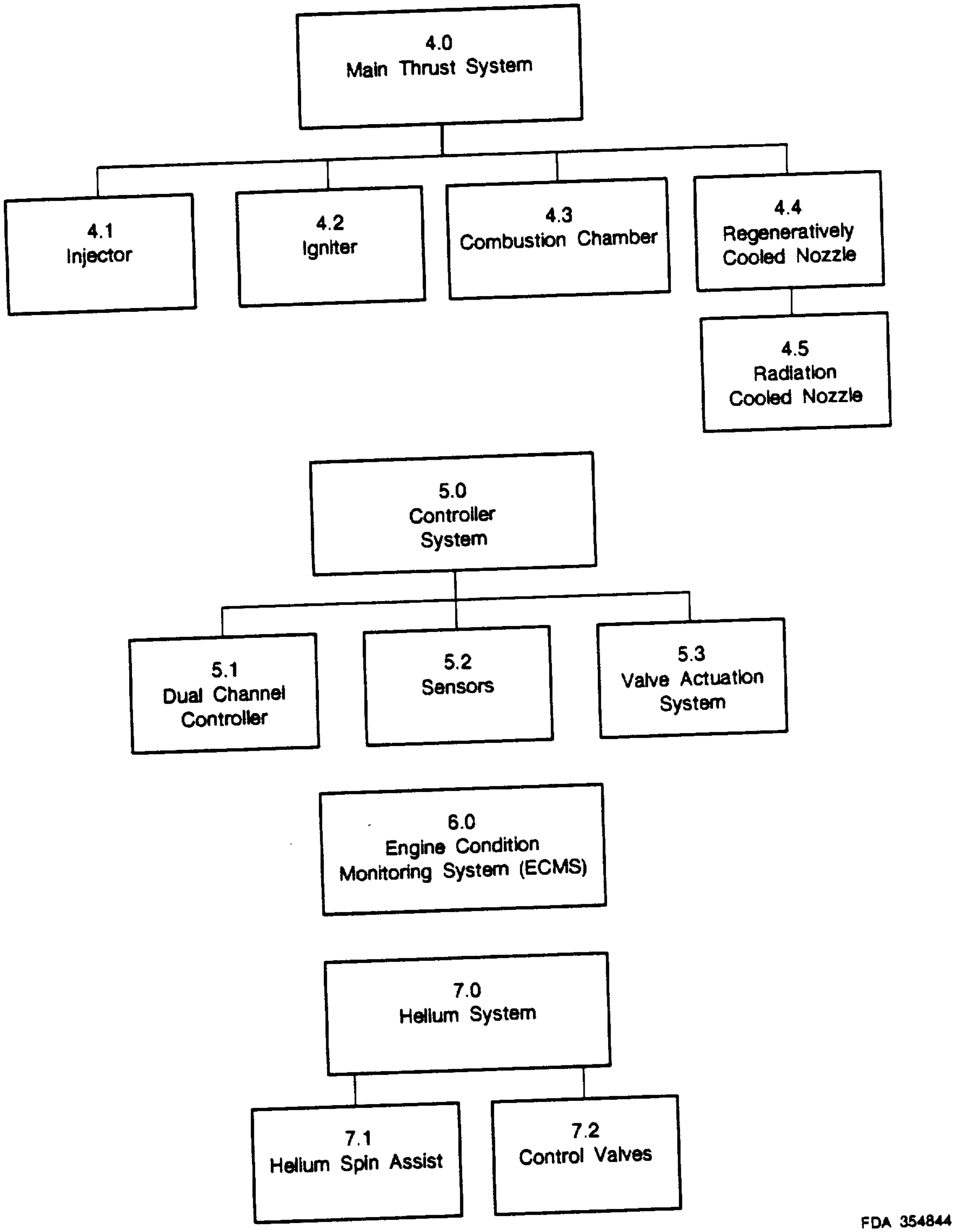

Figure 4.1.1.7-7. Preliminary FMEA — Main Thrust, Control, Engine Condition Monitoring, and Helium Systems (Sheet 4 of 4)

Table 4.1.1.7-1. Failure Mode and Effects Analysis

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Gas Generator Item 1.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 1 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
1.0 GAS GENERATOR Fuel and oxygen are injected, atomized, mixed and burned in the gas generator, creating an expanding gas mixture which is used to power both the fuel and oxygen turbopumps.				
1.1 INJECTOR Meters and injects the fuel and/oxygen flow into the gas generator combustion chamber, causing atomization and mixing for efficient combustion.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or material defect.	1. Premature mixing of fuel and LOX could lead to an explosion.	I		Structural margins of 1.5 inherent in the design should preclude this type of failure. Elimination of braze and weld will preclude premature mixing failures.
Failure mode: 2. Loss of liner integrity Cause: 2. Loss of cooling to liner	2. Burn through of injector liner could fire.	I		Design of Rigimesh cooling and structural margins should preclude problems.
Failure mode: 3. Injector burn through Cause: 3. SPUD misaligned/broken or localized clogging of face plate rigimesh.	3. Possible fire.	I		Design of Rigimesh cooling and structural margins should preclude problems.
1.2 IGNITER A series of sparks across the igniter plug gap and the addition of the proper igniter propellant flow creates a continuous torch which lights and keeps lit the chamber propellants.				
Failure mode: 1. Shorting Cause: 1. Cracked pressure insulation	1. No effect on normal engine operation. Engine unable to start or restart.	IIIR		Redundant spark igniters are provided
Failure mode: 2. Loss of electrical Cause: 2. Circuit malfunction	2. Same	IIIR		Redundant electrical leads are provided.
Failure mode: 3. Loss of propellant flow Cause: 3. Blockage of passage.	3. No ignition flame available.	II		Filters provided in lines.

Table 4.1.1.7-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Gas Generator Item 1.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
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Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
1.3 COMBUSTION CHAMBER				
Atomizing, mixing and burning of the gas generator propellants takes place here. Expanding gas then exits to the fuel turbine.				
Failure mode: 1. Non-uniform temperature profile. Cause: 1. Malfunction of the injector.	1. Combustion chamber burn through could lead to engine shutdown.	II		Automatic shutdown would be controlled by redline limits.
Failure mode: 2. Loss of propellant flow Cause: 2. Plugged propellant passages.	2. Fuel line blockage leads to LOX rich and over temp with possible explosion. LOX plug would lead to partial loss of engine performance, and excessive plugging would lead to engine shutdown.	I		Design features adequate filters, and the numbers of injectors should preclude loosing enough to create a problem.
1.4 GAS GENERATOR EXHAUST				
Vents the hot Gas Generator exhaust from the LOX turbopump to the nozzle.				
Failure mode: Fatigue Cause: Vibration, thermal growth, or material defect.	Loss of cooling film to the radiation cooled nozzle. Possible loss of engine performance.	II		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems.

Table 4.1.1.7-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Fuel System Item 2.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 3 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
2.0 FUEL SYSTEM Provides adequate pressure fuel for engine cooling requirements and gas generator and main thrust chamber injection using expanding gas from the gas generator to power the turbine. 2.1 HIGH PRESSURE FUEL TURBOPUMP (HPFTP) Provides high pressurize liquid fuel to satisfy engine fuel and cooling requirements. Failure mode: 2.1.1 TURBINE Uses expanding gas from the gas generator to create rotary motion to power the fuel pump. Failure mode: 1. Loss of turbine drive Cause: 1. Turbine blade or vane failure from fatigue.	1. Pump shutdown leading to engine shutdown.	II		Built in containment precludes impact to surrounding area. Use of advanced materials and cooling techniques provide enhanced turbine durability.

Table 4.1.1.7-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Fuel System Item 2.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 4 of 15
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
2.1.2 FUEL PUMP				
Pressurizes fuel for delivery to: 1. The main thrust chamber and the gas generator.				
Failure mode: 1. Ruptured housing and exit airfoil (splitter) Cause: 1. Excessive load or material defect	1. Could lead to explosion	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
Failure mode: 2. Excessive fuel seal leakage Cause: 2. Wear or sticking of the seal.	2. Overboard vent and loss of engine performance and shutdown.	II		Close clearance K.E. seal technology control leakage to an acceptable level. Vent line pressure monitors severity to preclude shutdown.
Failure mode: 3. Leakage through flange joint. Cause: 3. Seal, bolt, or flange failure such as fatigue.	3. Same as 1.	I		Same as 1.
Failure mode: 4. Bearing failure Cause: 4. Loss of cooling, excessive load, or defect.	4. Loss of bearing could lead to explosion.	I		Advanced bearing technology and controlled cooling coupled with inspection should preclude failures.
Failure mode: 5. Cavitation Cause: 5. Reduced cooling capability. If severe enough could cause imbalance and bearing failure.	5. Loss of bearing could lead to explosion.	I		Advanced bearing technology and controlled cooling coupled with inspection should preclude failures.
2.2 FUEL VALVES/PLUMBING				
Delivers and controls fuel flow from the tank to both the gas generator and thrust chamber injectors.				
2.2.1 FUEL GAS GENERATOR CONTROL VALVE (FGCV)				
The FGCV is a variable valve located downstream of the fuel Turbopump and upstream of the gas generator to control fuel flow into the gas generator which in turn controls the HPFTP and HPOTP turbine speeds.				
Failure mode: 1. Fails full open Cause: 1. Contamination, wear, loss of signal, or vibration.	1. Uncontrollable increased engine thrust. Could require engine shutdown.	IIR		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. Mechanical ganging with the OGCV requires both valves to fail or a loss of linkage between the valves.

Table 4.1.1.7-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Fuel System Item 2.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 5 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
Failure mode: 2. Fails closed Cause: 2. Contamination, wear,loss of signal, or vibration.	2. LOX rich mixture could cause GG chamber burn through and fire.	1R		Automatic shutdown would be controlled by redline limits. Mechanical ganging with the OGCV requires both valves to fail or a loss of linkage between the valves.

Table 4.1.1.7-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Fuel System Item 2.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 6 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
2.2.2 <u>FUEL SHUT OFF VALVE (FSOV)</u> Located downstream of fuel turbopump. Strictly an on/off functioning valve. Valve closed during engine pre-start cooldown, open during engine run, and closed at shutdown.				
<u>Failure mode</u>: 1. Fails open <u>Cause</u>: 1. Contamination, wear,loss of signal, or vibration.	1. Normal position at normal power level. No effect on engine operation.	III		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
<u>Failure mode</u>: 2. Fails closed <u>Cause</u>: 1. Contamination, wear,loss of signal, or vibration.	2. LOX rich combustion leading to main combustion chamber burn through resulting in possible fire and engine shutdown.	I		Automatic shutdown would be controlled by redline limits.
2.2.3 <u>FUEL PLUMBING</u> Plumbing transports the various propellants to and from the valves to the designated areas.				
<u>Failure mode</u>: Fatigue <u>Cause</u>: Vibration, thermal growth, or material defect.	Possible fire.	I		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems. Proper bracketry provided to dampen line vibration.
2.2.4 <u>FUEL PLUMBING GIMBAL</u> Provides mechanical interface between the HPFTP inlet with the vehicle main fuel supply. Permits engine vectoring without distortion of the fuel plumbing or disrupting fuel flow.				
<u>Failure mode</u>: Fatigue <u>Cause</u>: Vibration, thermal growth, or material defect.	Possible fuel leak and fire.	I		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems. DVS testing to assure proper design margins.

Table 4.1.1.7-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Oxidizer System Item 3.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 7 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
3.0 OXIDIZER SYSTEM Provides oxygen at an adequate pressure to enter the main oxygen pump. 3.1 HIGH PRESSURE OXIDIZER TURBOPUMP (HPOTP) Provides high pressure oxygen to be injected and combusted in the main thrust chamber and gas generator. 3.1.1 TURBINE Uses expanding gas from the gas generator to create rotary motion to power the oxygen pump.				
Failure mode: 1. Loss of turbine drive Cause: 1. Turbine blade or vane failure from fatigue.	1. Pump shutdown leading to engine shutdown.	II		Built in containment precludes impact to surrounding area. Use of advanced materials and cooling techniques provide enhanced turbine durability.
3.1.2 OXYGEN PUMP Pressurizes liquid oxygen for delivery to: 1. O2 gas generator heat exchanger (to pressurize O2 tank). 2. The main thrust chamber, and 3. the gas generator.				
Failure mode: 1. Ruptured housing Cause: 1. Excessive load or material defect	1. Could lead to explosion	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
Failure mode: 2. Leakage through flange joint. Cause: 2. Seal, bolt, or flange failure such as fatigue.	2. Same as 1.	I		Same as 1.
Failure mode: 3. Bearing failure Cause: 3. Loss of cooling, excessive load, or defect.	3. Loss of bearing could lead to explosion.	I		Advanced bearing technology and controlled cooling coupled with inspection should preclude failures.

Table 4.1.1.7-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Oxidizer System Item 3.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
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Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
3.2 GOX HEAT EXCHANGER The GOX heat exchanger uses the O2 pump turbine exhaust to vaporize a portion of the O2 pump discharge flow to pressurize the oxygen tank.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or defect	1. Possible leakage of gaseous oxygen which may cause an explosion.	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
Failure mode: 2. Plug Cause: 2. Contamination	2. Loss of tank pressure supply. Reduced LOX inlet pressure may cause loss of engine performance.	II		Design of HEX precludes any mixing of fuel and GOX due to structural failure. Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
3.3 POGO ACCUMULATOR The POGO Accumulator eliminates transmission of low frequency flow oscillations in to the high pressure oxidizer turbopump which in turn eliminates main combustion chamber pressure oscillations.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or defect	1. Possible leakage of gaseous oxygen which may cause an explosion.	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature. DVS testing to assure proper design margins.
3.4 OXIDIZER VALVES/PLUMBING Delivers and controls the oxygen flow from the tank to the gas generator and main thrust chamber injectors.				

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Table 4.1.1.7-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Oxidizer System Item 3.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 9 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
3.4.1 MAIN OXIDIZER VALVE (MOV)				
The MOV is a dual purpose valve located between the oxygen turbopump and the main chamber injector. The MOV serves as an on/off valve during start and shutdown and as a scheduled control valve during engine operation.				
Failure mode: 1. Fails open Cause: 1. Contamination, wear, loss of signal, or vibration.	1. Normal position at normal power level. No effect on engine operation.	III		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
Failure mode: 2. Fails closed Cause: 2. Same as above.	2. Abnormal engine shutdown	II		Same as above
3.4.2 OXIDIZER GAS GENERATOR CONTROL VALVE (OGCV)				
The OGCV is a dual purpose sleeve type valve located upstream of the gas generator and downstream of the oxygen pump discharge. The valve serves as an on/off valve during start and shutdown, and as a closed loop thrust control during engine operation.				
Failure mode: 1. Fails open Cause: 1. Contamination, wear, or vibration.l defect	1. Uncontrollable increased engine thrust. Could require engine shutdown.	IIR		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. Mechanical ganging with the FGCV requires both valves to fail or a loss of linkage between the valves. DVS and bench qualification testing to assure proper design margins.
Failure mode: 2. Fails closed Cause: 2. Contamination, wear, or vibration.l defect	2. Abnormal engine shutdown.	II		2. Same as above
3.4.3 OXYGEN PLUMBING				
Plumbing transports the various propellants to and from the valves to the designated areas.				
Failure mode: Fatigue Cause: Vibration, thermal growth, or material defect.	Possible explosion	I		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems.
3.4.4 OXYGEN PLUMBING GIMBAL				
Provides mechanical interface between the				

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Table 4.1.1.7-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Oxidizer System Item 3.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 10 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
HPOTP inlet with the vehicle oxygen supply. Permits engine vectoring without distortion of the oxygen plumbing or disrupting fuel flow. Failure mode: Fatigue Cause: Vibration, thermal growth, or material defect.	Possible LOX leak and explosion.	1		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems. DVS testing to assure proper design margins.

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Table 4.1.1.7-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Main Thrust System Item 4.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 11 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
4.0 MAIN THRUST SYSTEM Injects, atomizes, mixes, and burns fuel and oxygen to form hot gaseous reaction products, which are accelerated through the chamber throat and expanded through the nozzles to be ejected at a high velocity and create thrust.				
4.1 IGNITER A series of sparks across the igniter plug gap and the addition of the proper igniter propellant flow creates a continuous torch which lights and keeps lit the chamber propellants.				
Failure mode: 1. Shorting Cause: 1. Cracked pressure insulation	1. No effect on normal engine operation. Engine unable to start or restart.	IIR		Redundant spark igniters are provided
Failure mode: 2. Loss of electrical Cause: 2. Circuit malfunction	2. Same	IIR		Redundant electrical leads are provided.
Failure mode: 3. Loss of propellant flow Cause: 3. Blockage of passage.	3. No ignition flame available.	II		Filters provided in lines.
4.2 INJECTOR Meters and injects the chamber propellants into the combustion chamber, causing atomization and mixing for efficient combustion.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or material defect.	1. Premature mixing of fuel and LOX could lead to an explosion.	I		Structural margins of 1.5 inherent in the design should preclude this type of failure. Elimination of braze and weld will preclude premature mixing failures. DVS testing to assure proper design margins.
Failure mode: 2. Loss of liner integrity Cause: 2. Loss of cooling to liner	2. Burn through of injector liner and possible fire.	I		Design of Rigimesh cooling and structural margins should preclude problems.
Failure mode: 3. Injector burn through Cause: 3. SPUD misaligned/broken or localized clogging of face plate rigimesh.	3. Possible fire.	I		Design of Rigimesh cooling and structural margins should preclude problems.

Table 4.1.1.7-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Main Thrust System Item 4.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 12 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
4.3 COMBUSTION CHAMBER Provides a chamber for the mixing, atomization, burning and partial expansion of the chamber propellants, accelerating the hot gases to an area ratio of 4:1. Provides a heat exchanger for delivery of hot fuel to the gas generator and engine turbomachinery.				
Failure mode: 1. Non-uniform temperature profile. Cause: 1. Malfunction of the injector.	1. Combustion chamber burn through could lead to engine shutdown.	II		Automatic shutdown would be controlled by redline limits.
Failure mode: 2. Loss of propellant flow Cause: 2. Plugged propellant passages.	2. Fuel line blockage leads to LOX rich and over temp with possible explosion. LOX plug would lead to partial loss of engine performance, and excessive plugging would lead to engine shutdown.	I, II		Design features adequate filters, and the numbers of injectors should preclude loosing enough to create a problem.
4.4 REGENERATELY COOLED NOZZLE Primary purpose is to increase engine performance by expanding the combustion chamber gases from an area ratio of 4:1 to an area ratio of 31:1. Also acts as a heat exchanger to prepare fuel for chamber injection.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or material defect.	1. Leakage of fuel external to nozzle could lead to fire and LOX rich combustion burn through in chamber leading to engine shutdown.	I		Structural margins of 1.5 inherent in the design should preclude this type of failure. DVS testing to assure proper design margins.
Failure mode: 2. Loss of propellant due to tube failure Cause: 2. Vibration, thermal growth, or material defect.	LOX rich combustion leading to chamber burn through resulting in possible fire or engine shutdown.	I		Automatic shutdown controlled by Redline limits.
4.5 RADIATION COOLED NOZZLE Increases engine performance by expanding the combustion chamber gases from an area ratio of 31:1 to 55:1.				
Failure mode: Structural failure Cause: Vibration thermal growth, or material defect.	Loss of engine performance.	II		Design margin of 1.5 should preclude this type of problem. DVS testing to assure proper design margins.

Table 4.1.1.7-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Control System Item 5.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 13 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
5.0 CONTROLLER SYSTEM Provides a self-contained system for engine control. System includes and electronic controller, engine sensors, actuation devices, valves, and spark igniters. Failure mode: Cause:				
5.1 DUAL CHANNEL CONTROLLER Provides engine control and monitoring using input from sensing devices and output to actuation devices. Failure mode: Loss of multiplexing capability converting sensor analog or frequency signals to digital signals. Cause: Internal failure of converter circuit.	No effect on normal engine operation.	III		Dual redundant controller allows normal operation after the first failure.
5.2 SENSORS TBD Failure mode: Cause:				
5.3 VALVE ACTUATION SYSTEM TBD Failure mode: Cause:				

Table 4.1.1.7-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle
Functional Assy: Eng. Condition Monitor Sys. Item 6
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 14 of 15
Issue Date: Feb. 27,1989
Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
6.0 ENGINE CONDITION MONITORING SYSTEM (ECMS) TBD Failure mode: Cause:				

Table 4.1.1.7-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Gas Generator Cycle	**Prepared by:** R.L. Pugh	**Page:** 15 of 15
Functional Assy: Helium System Item 7.0	**Approved by:** W.E. Annas	**Issue Date:** Feb. 27,1989
		Rev. Date: Feb. 27, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criti-cality	Detection Method	Considerations
7.0 HELIUM SYSTEM Delivers and controls helium flow from the vehicle supply to be used for valve operation, emergency shutdown operation, turbopump spin assist, and control line purging. 7.1 HELIUM SPIN ASSIST TBD Failure mode: Cause: 7.2 HELIUM SYSTEM CONTROL SOLENOIDS TBD Failure mode: Cause:				

Results

This section is intentionally left blank at this time. It will be developed as the analysis proceeds.

Conclusions

This section is intentionally left blank at this time. It will be developed as the analysis proceeds.

Applicable Documents

NHB 5300.4(1D.2) Safety, Reliability and Quality Provisions for SSME Programs

Procedure

This report was prepared in accordance with STBE Reliability requirements.

Ground Rules and Assumptions

The following ground rules and assumptions were used in the preparation of the FMEA.

Level of Analysis

a. The analysis is conducted at the component and major subassembly level. In subsequent updates, the FMEA will contain both a hardware and a functional analysis. To show the distinction, the index numbers have been modified to differentiate between functional and hardware type of analysis.

b. Condition category I and II items will be analyzed to the level necessary to verify that adequate controls are in place.

c. External fire, explosion, or case penetration that could endanger the remaining engines are classified as Condition Classification I.

d. The worst case effect of leakage is fire/explosion. In this analysis, leakage will be classified as Condition I.

e. The analysis was conducted considering the engine operation at normal power. Subsequent updates will consider the mission phases in the following paragraph.

f. The helium solenoids, valve actuators, sensors, and monitoring devices have not been analyzed. Analysis of these items will be provided in subsequent updates.

Mission Phases

The engine will be analyzed for potential failure modes of a single flight in each of the following mission phases:

Event	*Phase*	*Abbreviation*
Pre-start	Pre-ignition	P
Start Command	Engine Start	S
Normal Power	Main Stage Operation	N
Max Thrust	Main Stage (Lift Off)	M
Cutoff Command	Shutdown	C
Dump	After Shutdown	D

Failure Modes

a. Failure modes will be identified for each level or output applicable to the operational phase being considered.

b. The analysis will consider only one failure mode occurring at any given time and will be the basis for establishing Condition Classification.

c. Leakage at all mechanical joints shall be analyzed.

d. Welded or brazed joints shall be analyzed for structural failure.

e. Failure mode causes shall be identified for all Condition Classifications I, II, and III.

Reaction Time

The analysis will determine the time for the failure effect to occur, and it is specified in units of time as indicated below:

	Definition	*Abbreviation*
Immediate	— Less than a second	IMM
Seconds	— 1 to 60 seconds	SEC
Minutes	— 60 seconds to 60 minutes	MIN
Hours	— 60 minutes to 24 hours	HRS
Days	— 24 hours to mission completion	DYS

If a failure detection method is available, it is specified with time to safely correct the problem. If a detection system is available, but would not safely correct the problem, this is also noted.

Failure Effects

a. Failure effects will be analyzed for each identified failure mode. Where a piece part failure can cause a failure of another part, the Condition Classification will be based on the likely effect of the resultant or combined failures.

b. Condition designation should reflect "the most likely" potential effect of the failure mode in either countdown or flight. This includes possible cata-

strophic effects, such as fire/explosion, as well as effects of loss of hardware functions. Single failures, such as leakage of LO_2, in the presence of a possible ignition source, will be listed as potential fire/single failure point. Leakage of hot gas is classified Condition I.

Structural Failure Modes

Structures are excluded from the FMEA, with the exemptions listed below.

a. Pressure vessels, component housings, ducts, fluid lines, sliding joints, expansion joints, bolts, attach fittings, or load carrying members such as rods will be analyzed for structural failures.

b. Structural failures of piece parts shall be considered valid failure causes for component failure mode analysis.

c. Items which have a single mechanical barrier between oxidizer and fuel/combustible gas.

d. Items that are known to develop "acceptable defects" within their allowed time for usage, shall be analyzed for worst case of defect propagation.

e. Aerodynamically sensitive items.

f. Items having internal cavities which can induce an internal overpressure from migrating fluid because of leak from inside or outside.

g. Leakage at all joints that are formed by weld or braze shall be analyzed to assess the effect of a leak impinging on other components or flammable surfaces.

h. Welds or braze joints that cannot be inspected will be analyzed for leakage and for structural failure effects.

Criticality Category

The criticality category for each failure mode will be assessed for its effect on mission as follows:

Condition	*Mission Effect*
I	A potential failure mode resulting in fire/explosion or other hazardous condition that could impact the surrounding area.
II	A potential failure mode that could result in an unscheduled safe engine shutdown.
III	A potential failure mode that could result in the engine safely operating outside of required parameters.

4.1.1.7.4.2 Maintainability

Preliminary maintainability design criteria for the STBE has been defined and provided to design engineers in a memorandum. The design criteria was derived from the statement of work

(SOW), preliminary guidance from ALS airframers, and experience gained from other liquid rocket programs. Experience gained includes the Pratt & Whitney RL10 and Alternate Turbopump Development (ATD) programs and information from various NASA reports relative to the SSME, F1, J2, H1, RS-27, Thor, and Atlas programs. Updates will be made to the maintainability design criteria as additional requirements are identified.

Pratt & Whitney maintainability engineering has been working in conjunction with the ALS airframers to define an overall maintenance concept for the STBE. Definition of the maintenance concept will provide necessary guidance in identifying those propulsion system components that are either line replaceable units (LRU's) or modules. The definition and lists of LRU's and modules and preliminary maintenance concepts will be provided in subsequent reports.

4.1.1.7.4.3 System Safety

To support the development of design requirements, System Safety developed Fault Tree Analyses of the gas generator engine systems and their major components. These Fault Trees are high-level models to study the overall systemic effects of "generic" events such as "turbopump mechanical malfunction". Detailed fault trees investigating events such as "bearing rate fracture" within a turbopump will be developed during Phase B studies.

The Fault Trees were analyzed to identify those events with possible catastrophic results. The identified events and their effects on the system were then analyzed to determine safety requirements which would eliminate or reduce the probability and/or severity of the undesired effect. These requirements have been summarized and provided to Project Engineering for inclusion as engine system design requirements during Phase B design activities.

The objective of this effort is to reduce the probability of a catastrophic engine event (one that results in the loss of the payload or the vehicle, or the death or serious injury of a person) to the lowest possible level. This will be accomplished by using the fault trees to identify those diagnostic elements which detect potentially hazardous conditions in time to effect an engine shutdown before the event becomes catastrophic. The overall goal is to contain the damage within the malfunctioning engine thus avoiding potential damage to an adjacent engine.

4.1.1.7.5 Engine Performance

The STBE derivative gas generator engine system performance was determined using the accepted JANNAF methodology. Vacuum specific impulse was calculated separately for both the main chamber nozzle and the GG nozzle. Overall engine performance was calculated by mass weighing the main chamber flow performance with the GG flow performance. Table 4.1.1.7-2 summarizes main chamber and GG performance parameters at the design thrust level of 644,898 pounds sea level.

During this study program, detailed aerothermal analyses were made to predict component performance levels. Results of these analyses were incorporated into a steady state power balance model of the complete engine. A simplified flow schematic is presented in Figure 4.1.1.7-8 with key operating parameters noted for the design thrust level. Table 4.1.1.7-3 defines performance of the individual components and their operating environments for the derivative engine at design power level.

Table 4.1.1.7-2. *STBE Derivative Gas Generator Engine Performance — Design Power Level*

	Design Power Level	
	Main Chamber	*Gas Generator*
Pressure — psia	2250	221.6
Mixture Ratio	3.48	0.301
Nozzle Area Ratio	28	5
Flow Rate — lb/sec	1993	185
Vacuum Thrust — lb	679922	31901
Vacuum I_{sp} — sec	342.9	172.4
	Overall Engine	
Vacuum Thrust — lb	711,823	
Vacuum Del. I_{sp} — sec	328.4	
S.L. Thrust — lb	644,898	
S.L. I_{sp} — sec	297.5	

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4.1.1.7.6 Engine Costs

This section summarizes cost estimates for the 645K SL thrust, 2250 psia chamber pressure, Derivative STBE Gas Generator cycle. Table 4.1.1.7-4 summarizes significant costs for the engine.

The DDT&E Cost includes all of the functions required to design, develop, test and evaluate the engine system. All of the DDT&E functions shown in the ALS engine WBS (see Volume III) have been included. Development Cost is based on a 90-month phase C/D program with 960 engine firings for the STME, and 488 for the Derivative STBE. Sufficient accountable firings have been included in the program to demonstrate 0.99 engine reliability with one failure.

The engine Theoretical First Unit (TFU) production cost includes all the recurring operational production cost elements specified in the ALS engine WBS. It includes manufacturing and acceptance of the Integrated Engine System, System Engineering and Integration, Program Management, Facilities Maintenance and Tooling Maintenance. The TFU estimate is based on a lot size of 100 and a 90-percent learning curve.

The Operations Cost per launch per engine includes all costs associated with the operational flight program as described in the ALS engine WBS. It includes Program Management, System Engineering and Integration, Facilities Maintenance, Operation and Support, and Training. The Operations Cost is based on a flight rate of 10 missions per year and it is the estimated cost that will be achieved after 100 total missions have been flown.

Table 4.1.1.7-5 shows program costs, based on a total production buy of 425 Derivative STBEs and 175 STMEs. This is the nominal procurement projected for the STME/Derivative STBE program. The total production cost includes all recurring and non-recurring costs. The total operations cost is for 300 total flights over a 25-year period using seven reusable Derivative STBEs on the Booster stage and three reusable STMEs on the core stage.

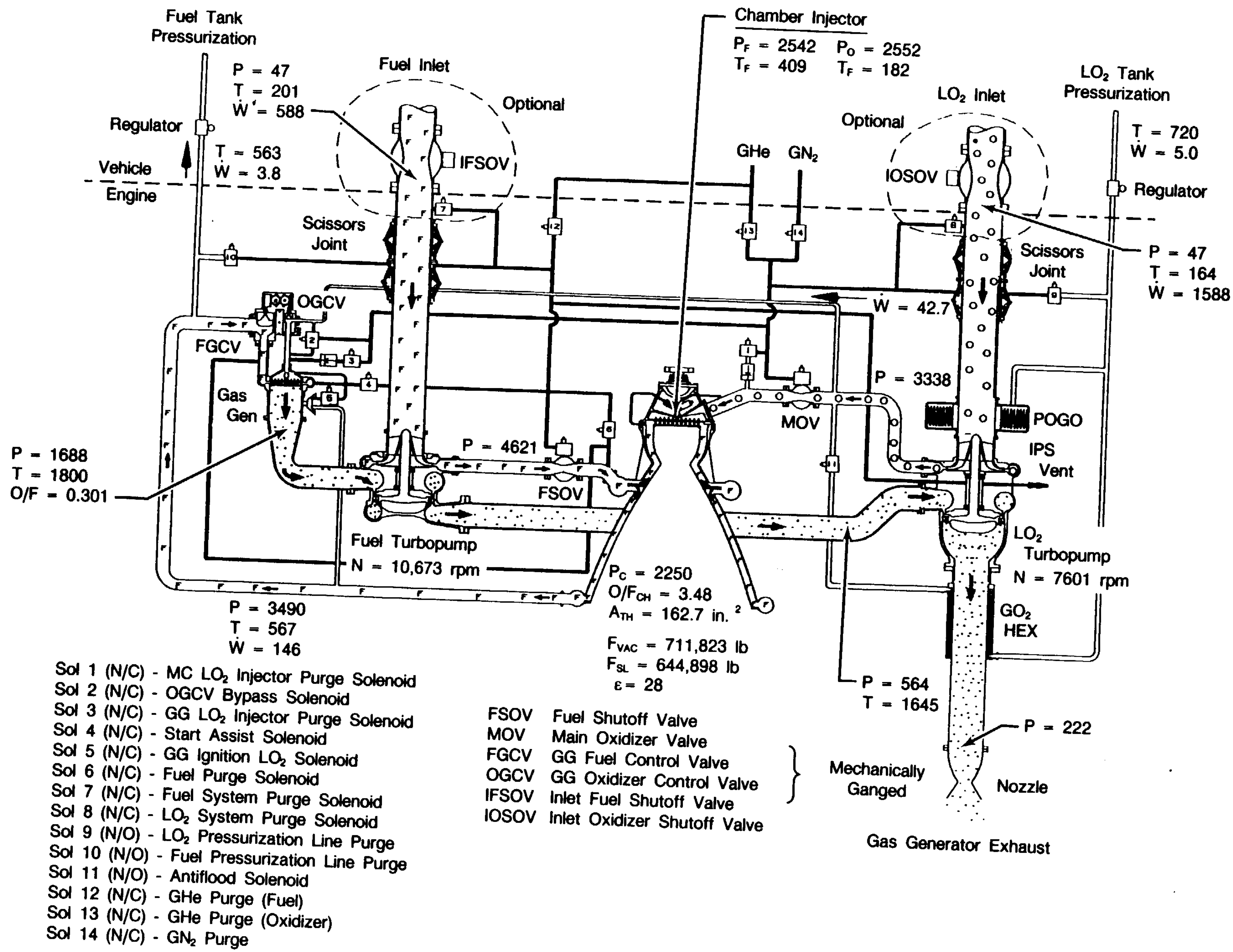

Figure 4.1.1.7-8. STBE Derivative Gas Generator Engine Operating Characteristics at DPL

Table 4.1.1.7-3. STBE Derivative Gas Generator Engine Performance — Design Power Level

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*****************************************
*            PRATT & WHITNEY            *
*  GAS GENERATOR CYCLE OFF-DESIGN DECK  *
*           STBE ENGINE STUDY           *
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ENGINE PERFORMANCE

VACUUM THRUST	711823.
SEA LEVEL THRUST	644898.
VACUUM IMPULSE	328.35
SEA LEVEL IMPULSE	297.48
TOTAL ENGINE INLET FLOW RATE	2176.6
OVERALL ENGINE MIXTURE RATIO	2.70

ENGINE HEAT TRANSFER

CHAMBER COOLANT DP	1806.
CHAMBER COOLANT DT	177.
CHAMBER Q	66563.
NOZZLE COOLANT DP	534.
NOZZLE COOLANT DT	333.
NOZZLE Q	41724.

CHAMBER PERFORMANCE

PRESSURE	2250.0
TEMPERATURE	6601.7
THRUST	679922.
IMPULSE	342.90
FLOW RATE	1982.9
THROAT AREA	162.71
NOZZLE AREA RATIO	28.
MIXTURE RATIO	3.48
NOZZLE EFFICIENCY	0.965
CSTAR EFFICIENCY	0.980

GAS GENERATOR PERFORMANCE

PRESSURE	1687.5
TEMPERATURE	1800.0
THRUST	31901.
IMPULSE	172.44
FLOW RATE	185.0
MIXTURE RATIO	0.301
NOZZLE EFFICIENCY	0.970
NOZZLE GAS CONSTANT	97.2
NOZZLE GAMMA	1.177
NOZZLE AREA	88.7

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	201.0	588.3	123.1	26.40
1ST STAGE EXIT	2325.1	216.0	588.3	145.4	26.55
MAIN PUMP EXIT	4621.3	230.3	588.3	167.2	26.74
FSOV INLET	4506.9	231.0	588.3	167.2	26.68
FSOV EXIT	4451.2	231.3	588.3	167.2	26.64
CHAM/COOL INLET	4369.8	231.8	442.2	167.2	26.60
CHAM/COOL EXIT	2563.5	408.8	442.2	317.8	15.79
CH INJ INLET	2542.0	408.6	442.2	317.8	15.75
NOZ/COOL INLET	4024.2	233.8	146.0	167.2	26.40
NOZ/COOL EXIT	3490.4	566.7	146.0	453.0	10.42
TANK PRESS OUT	3249.7	563.0	3.8	453.0	9.93
TANK PRESS IN	47.0	421.4	3.8	453.0	0.17
FGCV INLET	3249.7	563.0	142.3	453.0	9.93
FGCV EXIT	2385.1	545.3	142.3	453.0	7.91
GG INJ INLET	2276.1	542.4	142.3	453.0	7.63

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1588.3	61.6	71.17
MAIN PUMP EXIT	3338.3	178.8	1588.3	72.8	71.74
GOX HEX IN	3228.9	179.3	5.0	72.8	71.58
TANK PRESS IN	47.0	720.0	5.0	275.4	0.22
MOV INLET	3228.9	179.3	1540.6	72.8	71.58
MOV EXIT	2647.4	181.7	1540.6	72.8	70.70
CH INJ INLET	2552.5	182.0	1540.6	72.8	70.55
OGCV INLET	2880.3	180.7	42.7	72.8	71.05
OGCV EXIT	2668.2	181.6	42.7	72.8	70.73
GG INJ INLET	2434.7	182.5	42.7	72.8	70.37

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
FUEL TURB INLET	1532.7	1800.0	185.0
FUEL TURB EXIT	646.7	1655.5	185.0
LOX TURB INLET	563.9	1645.4	185.0
LOX TURB EXIT	289.1	1550.3	185.0
NOZZLE INLET PRES	221.6		

Table 4.1.1.7-3. STBE Derivative Gas Generator Engine Performance — Design Power Level (Continued)

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*************************************
*            PRATT & WHITNEY        *
* GAS GENERATOR CYCLE OFF-DESIGN DECK *
*          STBE ENGINE STUDY        *
*************************************
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PAGE 2

TURBOMACHINERY PERFORMANCE DATA

FUEL TURBINE

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.774	0.759
HORSEPOWER	19000.	17763.
SPEED (RPM)	10673.	10673.
S SPEED	35.2	44.3
S DIAMETER	1.84	1.54
MEAN DIAMETER (IN)	19.12	19.10
VEL.RATIO (ACTUAL)	0.47	0.48
MAX TIP SPEED	918.	941.
BLADE HEIGHT	0.58	1.10
AN SQUARED	39.7	75.2
EFFECTIVE AREA	14.07	21.17
PRES.RATIO (T/T)	1.54	1.54
GAS CONSTANT (FT)	97.20	
GAMMA	1.1626	

FUEL PUMP

	STAGE ONE	STAGE TWO
EFFICIENCY	0.713	0.729
HORSEPOWER	18570.	18192.
SPEED (RPM)	10673.	10673.
NPSH (FT)	177.7	12461.5
SS SPEED	27007.	1130.
S SPEED	910.	905.
HEAD (FT)	12373.	12405.
DIAMETER (IN)	18.28	18.28
TIP SPEED (FT/SEC)	852.	852.
VOL FLOW	10002.	9945.
HEAD COEF	0.5396	0.5410
FLOW COEF	0.0728	0.0724

LOX TURBINE

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.774	0.699
HORSEPOWER	12630.	12637.
SPEED (RPM)	7601.	7601.
S SPEED	50.9	55.4
S DIAMETER	1.16	1.06
MEAN DIAMETER (IN)	18.90	18.77
VEL.RATIO (ACTUAL)	0.40	0.40
MAX TIP SPEED	680.	715.
BLADE HEIGHT	1.60	2.77
AN SQUARED	54.9	94.4
EFFECTIVE AREA	38.67	50.39
PRES.RATIO (T/T)	1.36	1.43
GAS CONSTANT (FT)	97.05	
GAMMA	1.1697	

LOX PUMP

EFFICIENCY	0.756
HORSEPOWER	25267.
SPEED (RPM)	7601.
NPSH (FT)	62.4
SS SPEED	37052.
S SPEED	1037.
HEAD (FT)	6618.
DIAMETER (IN)	18.91
TIP SPEED (FT/SEC)	628.
VOL FLOW	10017.
HEAD COEF	0.5406
FLOW COEF	0.0821

VALVE DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL SHUT OFF VLV	55.7	22.83	588.3	1.24
FUEL GG VALVE	864.5	2.803	142.3	26.60
MAIN OXID VALVE	581.6	11.30	1540.6	18.01
LOX GG VALVE	212.0	0.521	42.7	7.36

INJECTOR DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL GG INJ	588.6	3.960	142.3	25.86
FUEL CH INJ	292.0	13.19	442.2	11.49
LOX GG INJ	747.2	0.279	42.7	30.69
LOX CH INJ	302.5	15.78	1540.6	11.85

Table 4.1.1.7-4. Derivative STBE Gas Generator Costs

Total Development Cost (DDT&E), M$658*
Production Cost (TFU), M$10.3
Operations Cost/Launch/Engine, M$0.142**
Constant FY87$

*Applies to Derivative STBE, an additional M$1183 Development Program is estimated for the STME.
**Based on the 100th mission, 10 missions per year, and seven boosters.

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Table 4.1.1.7-5. Total Program Cost for Combined STME/Derivative STBE Program

DDT&E, M$1841
Operational Production, M$3593
Operations, M$480
Product Improvement and Support Program, M$739
Total Program Cost, M$6653
Constant FY87$, Nominal Flight Schedule

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4.1.1.7.7 Engine/Vehicle Interface Requirements

All engine physical interfaces meet ALS ICD specifications. The fuel and oxidizer inlet ducts are configured on a 180-degree spacing and are 34 and 33 inches from the gimbal centerline respectively. The engine assembly could be converted to a 90-degree pump inlet spacing if a benefit to the vehicle is found to exist. A review of the vehicle contractors current vehicle cluster configurations indicates better access to the turbopumps when installed on the vehicle. As the engine maintenance concept evolves, module and LRU location of the engine assembly will be reviewed. Currently, hydrodynamic design has assumed that the inlet ducts are free of bends and are the same diameter as the pump inlet for at least five pipe diameters upstream of the inducer. As vehicle configurations stabilize, the sensitivity of the pump designs to inlet flow perturbations will be more fully addressed.

In addition to the propellant inlets, four additional fluid interfaces exist on the baseline engines: the two propellant tank pressurization flows, a nitrogen and a helium supply for engine purges. SSME interface locations were used for these fluid interfaces on current baseline ALS engine designs. Significant flexibility in the location of these lines exists to respond to vehicle requirements.

Nitrogen is required only during ground purges. Helium is required for the engine start system and for inflight purges and post shutdown purges. For those engine recovery concepts which involve sea recovery, an additional purge of the turbopump turbine cavities and bearing compartments prior to water impact through shipboard recovery is required to prevent corrosive sea air from being drawn into the turbopumps as the hot turbine structures cool. Vehicle considerations will likely guide the decision to use either nitrogen or helium for this purge. Additional refinement and quantification of the turbopump cavity volumes are required to quantify the flowrate requirements for all purges.

The proposed method of supplying vehicle electrical power is a vehicle mounted generator coupled to an auxiliary turbine driven by the fuel tank pressurization flow. Pressure drop across

the generator turbine lowers fuel tank pressurization flow to the 500 psi level downstream of the turbine. A conceptual design has been completed which would supply 25 kW DC power per engine, or 75 kW total in a three-power engine cluster. Growth margin exists to increase the 25 kW level if vehicle requirements increase. This concept removes the generator from the engine assembly to reduce gimbaled mass, lowering actuator loads. This approach is attractive since it is compact and does not require a separate hydrazine APU system as on the shuttle. Use of this system would require the tank pressurization flow to be continuous, not pulsed. If hydraulically operated thrust vectoring actuators are selected, an electrically driven hydraulic pump would be required in conjunction with this system.

Until ALS ICD limits are established governing leakage levels for propellants external of the engine, SSME leakage levels have been assumed. These levels would be appropriate if an enclosed propulsion compartment is selected. The specific flange design requirements derived from this leakage level include:

- Maximum leakage: 0.0001 sccs of helium at operating pressure
- Durability: 400 pressure cycles.

In tests conducted under the XLR-129 program, nineteen configurations of eight basic seal designs were tested using GN_2 under pressures between 50 and 7000 psi for 500 cycles. Under the SSME-ATD turbopump program additional high pressure seal tests have been performed to access the performance of lower cost nickel seal platings in place of silver and gold platings on E and C seal configurations.

Based upon review of these tests, these leakage requirements appear consistently obtainable. The resulting baseline ALS engine designs utilize Dynatube type fluid connectors on all lines one inch in diameter or less. The Dynatube connector provides the lowest cost and is the lightest design for small diameter lines. It has years of field experience and many hours of rig testing in aerospace applications to substantiate its performance. From an assembly standpoint, the Dynatube has an integral multi-useable seal to ensure that the seals are not omitted and provides more repeatable performance.

As line size increases past the one-inch diameter limit, assembly torque requirements become excessive for Dynatube and other tube type connectors. As a result, larger lines are joined using flanged connectors. The superior strength of the flanged connector makes it less likely to be damaged due to engine vibration. Due to the weight of large flanges, additional trade studies need to be performed to investigate such things as integral valves in line to reduce the total number of flanges and to minimize weight penalties as well as potential leakage sites.

4.1.1.7.8 *Preliminary Interface Control Document*

The following Interface Control Document for the STBE is submitted in its entirety with its unique paragraph, figure, table, and page numbering.

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PRELIMINARY INTERFACE CONTROL DOCUMENT FOR THE SPACE TRANSPORTATION BOOSTER ENGINE

ISSUE NO.: 3 DATE: March 31, 1989

VEHICLE APPLICATION: Space Transportation System

ENGINE IDENTIFICATION: TBD

ICD NO.: TBD

This document identifies the performance and physical interfaces or characteristics of the Space Transportation Booster Engine (STBE) that NASA and Pratt & Whitney agree to for designing the Space Transportation System and the STBE. This is the basis for specification requirements for the STBE.

Approved by ______________________________
NASA

Approved by ______________________________
Pratt & Whitney

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CONTENTS

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PRELIMINARY INTERFACE CONTROL DOCUMENT FOR THE SPACE TRANSPORTATION BOOSTER ENGINE

1.0 INTRODUCTION

1.1 SCOPE

The purpose of this Interface Control Document (ICD) is to define the preliminary engine/vehicle interface requirements as well as operational requirements for the Space Transportation Booster Engine (STBE).

1.2 ENGINE DEFINITION

The STBE is a liquid bipropellant rocket engine using liquid methane as fuel and liquid oxygen as oxidizer. The engine design uses the gas generator power cycle. The engine will produce a normal sea-level thrust of 644,898 pounds at a delivered specific impulse of 297.5 seconds.

2.0 ABBREVIATIONS AND SYMBOLS

AMP	Amperes
Deg	Degrees of angle or of temperature
°F	Degrees Fahrenheit
FT	Feet
GCH_4	Gaseous methane
GN_2	Gaseous nitrogen
GO_2 or GOX	Gaseous oxygen
He	Helium
HR	Hour
Hz	Hertz
I_{sp}	Specific impulse
lb or lbs	pound(s)
LCH_4	Liquid methane
LO_2	Liquid oxygen
LRU	Line replaceable unit
Max	Maximum
Min	Minimum or minute
MS	Millisecond
NPL	Normal power level
NPSP	Net positive suction pressure
O/F	Oxidizer to fuel ratio
Psia	Pounds per square inch, absolute
Psig	Pound per square inch, gage
°R	Degrees Rankine
SL	Sea level
Sec	Seconds
STBE	Space Transportation Booster Engine
TBD	To be determined
TVC	Thrust Vector Control
Vac	Vacuum

3.0 DOCUMENTS

3.1 DRAWINGS

The layout for the Space Transportation Booster Engine (STBE) is shown in Figure F-3.1. Detailed drawings of interface connections are TBD.

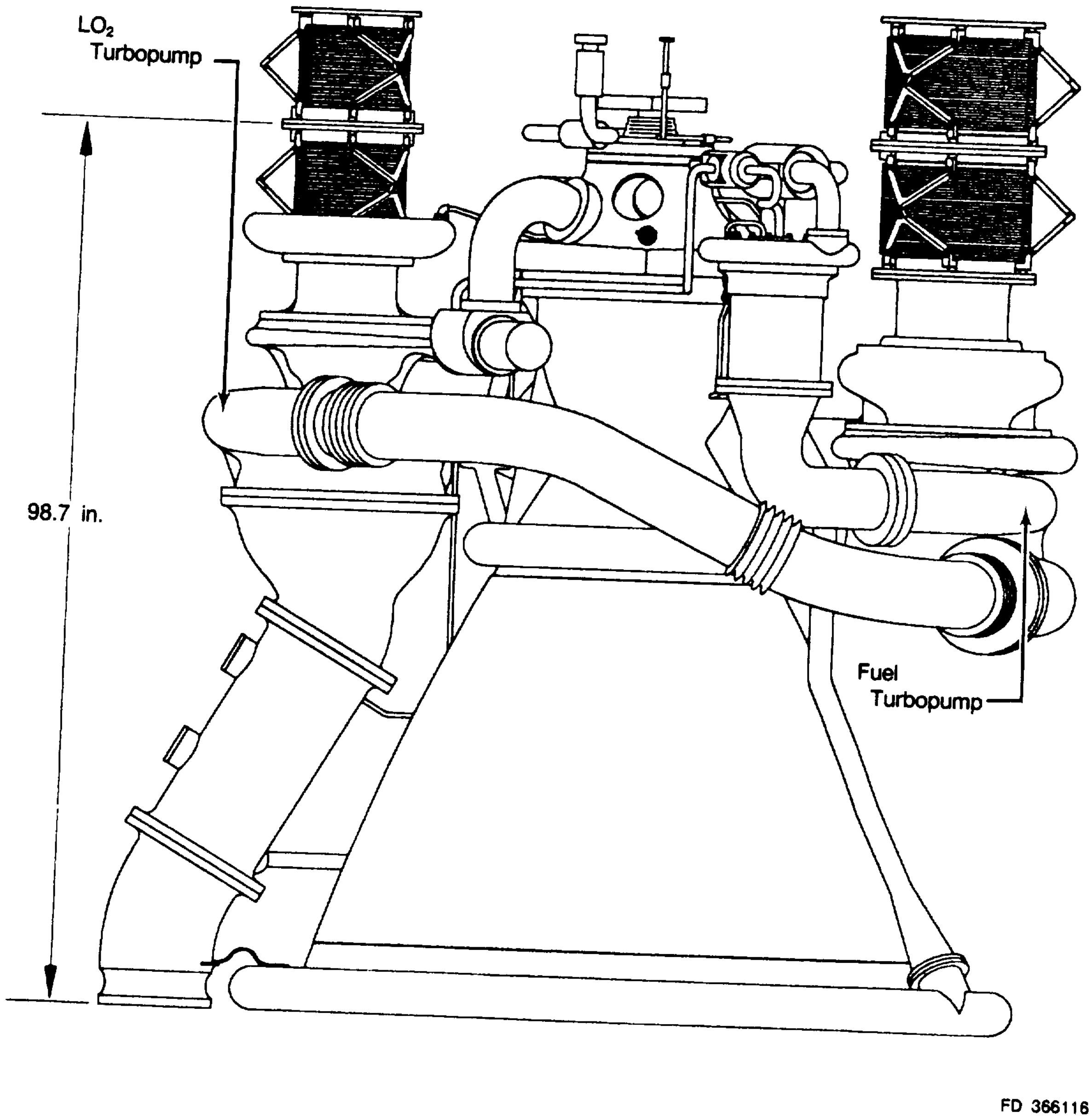

Figure F-3.1. STBE Layout Drawing (Sheet 1 of 3)

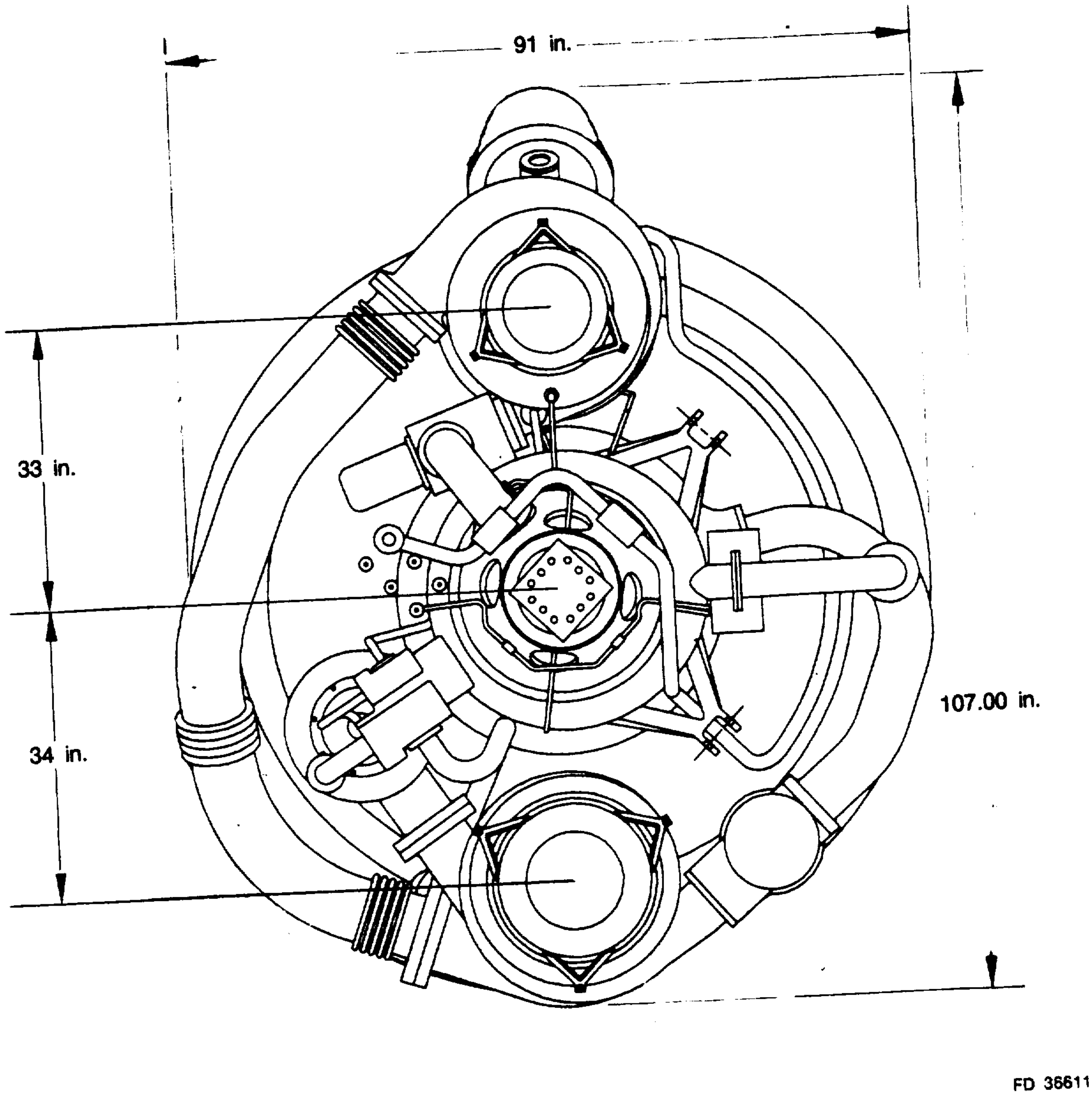

Figure F-3.1a. STBE Layout Drawing (Sheet 2 of 3)

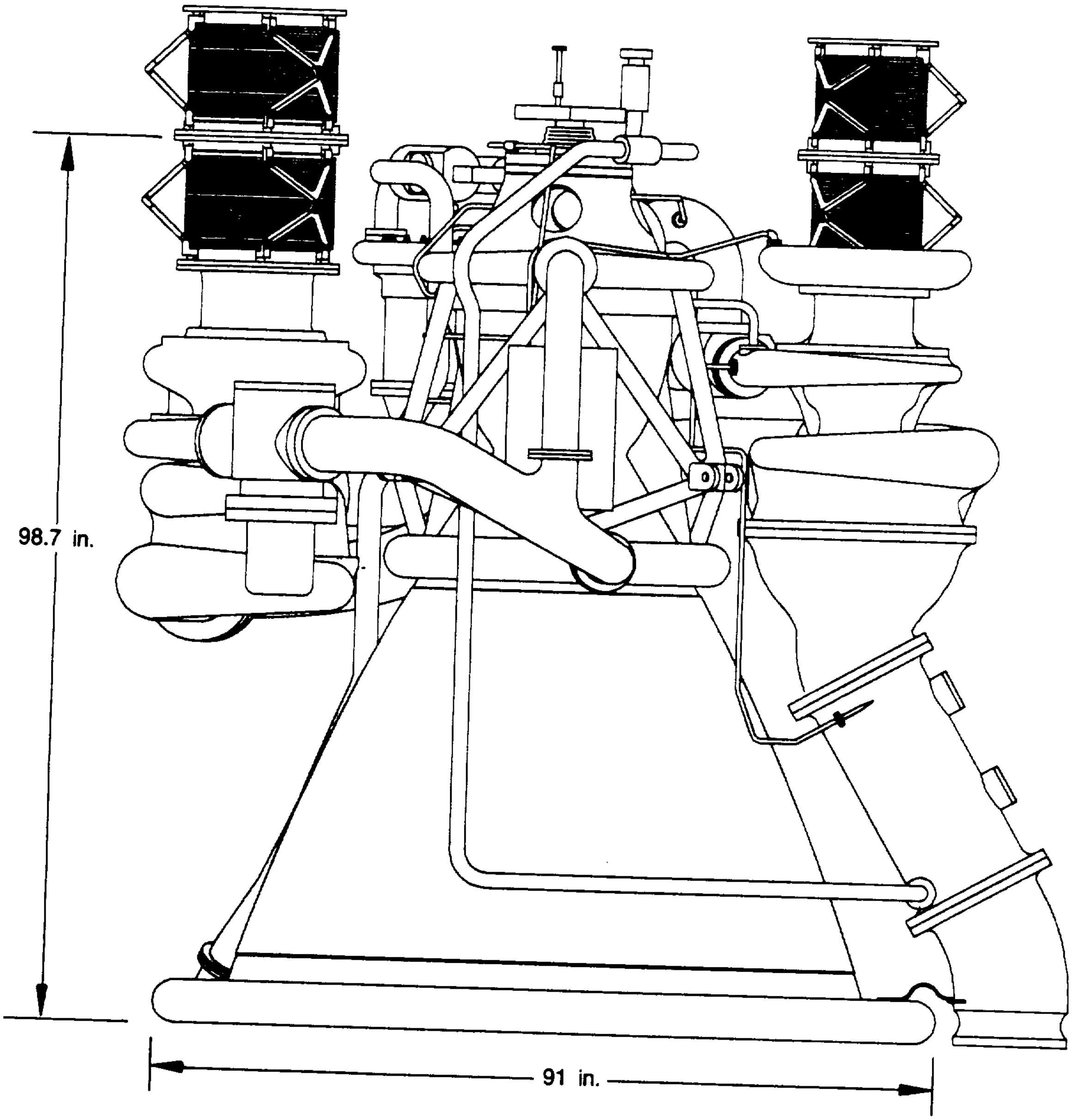

Figure F-3.1b. STBE Layout Drawing (Sheet 3 of 3)

3.2 APPLICABLE SPECIFICATIONS AND STANDARDS

Document	*Subject*
MIL-P-25508E-3	Propellant, Oxygen — Type II, Grade A or Equivalent.
MIL-P-27407A	Helium, Type I, Grade A (Gaseous) or Equivalent.
MIL-P-27401C	Nitrogen, Propellant Pressurizing Agent.
TBD	Propellant, Methane.
MIL-STD-704	Electric Power, Aircraft.

4.0 MASS CHARACTERISTICS

4.1 ENGINE WEIGHT

The estimated dry and wet weights of the Space Transportation Booster Engine (STBE) are provided in Table T-4.1.

Table T-4.1. *Estimated STBE Engine and Propellant Weights*

Basic Dry Weight (lb)	*Propellant Weight (lb)*	*Total Wet Weight (lb)*
6960	809	7769

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Thrust vector control actuators are not included in engine weights.

4.2 CENTER OF GRAVITY

The engine center of gravity is at the engine coordinates provided in Table T-4.2. The engine coordinate system is defined in Figure F-5.1.

Table T-4.2. *STBE Center of Gravity Coordinates*

Coordinate Axis	*Distance (in.)*
X	TBD
Y	TBD
Z	TBD

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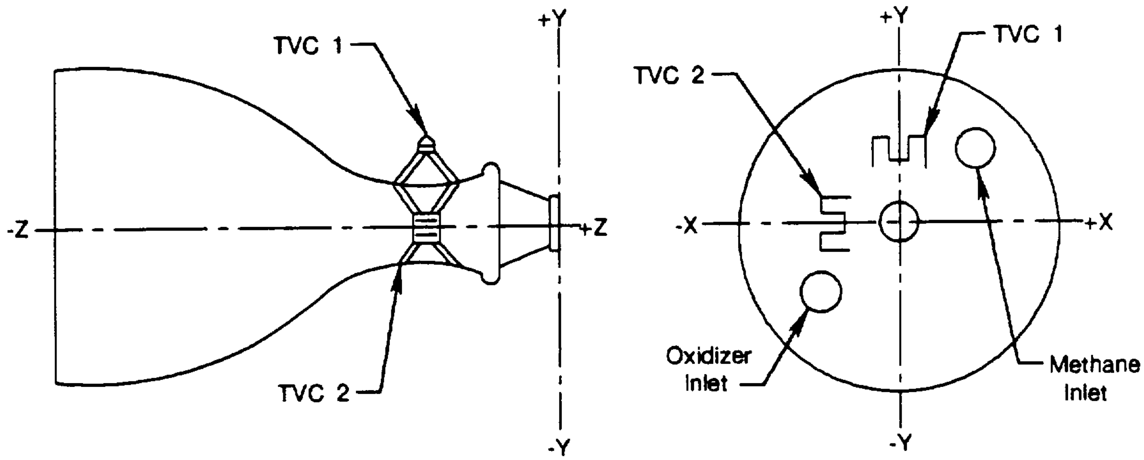
4.3 GIMBAL MOMENTS OF INERTIA

TBD



	X	Y	Z
Oxidizer Inlet	-23.3	-23.3	0
Methane Inlet	24.0	24.0	0
TVC Attach Point 1	0	25	-46
TVC Attach Point 2	-25	0	-46

FDA 366132

Figure F-5.1. Position of Propellant Inlet Ducts and Thrust Vector Control Attach Points Relative to Engine Coordinate Axis

5.0 ENGINE-TO-VEHICLE INTERFACES

5.1 ENGINE ENVELOPE AND SPACING REQUIREMENTS

5.1.1 Engine Static Envelope

Maximum overall dimensions for the Space Transportation Booster Engine (STBE) static envelope are as follows:

Max Diameter, in.	— 91, 107
Max Length, in.	— 99

These dimensions are indicated on Figure F-3.1.

5.1.2 Engine Dynamic Envelope

Maximum dimensions for engine dynamic envelope are shown on Figure TBD. This envelope represents volume required for an engine at maximum gimbal angle.

5.1.3 Engine-to-Engine Centerline Spacing

Required centerline spacing between engines is shown in Figure TBD. This spacing represents the distance between engines required for engine installation or removal, component removal and replacement, engine checkout and maintenance, and for clearance at maximum gimbal angle.

5.2 ENGINE-TO-VEHICLE PHYSICAL INTERFACES

The index of physical interfaces is shown in Table TBD. The interface configuration and location of these mechanical, fluid, and electrical connections is shown on Figure TBD, and discussed in subsequent paragraphs.

5.2.1 Gimbal Mount and Actuator Attach

The gimbal mount is the primary engine attachment to the vehicle and provides capability to gimbal the engine through the two actuator attach points on the engine, located 90 degrees apart. The gimbal actuator attachment positions are shown in Figure F-5.1.

5.2.2 Engine-to-Vehicle Alignment

The engine centerline shall be within TBD minutes of arc to the vehicle reference centerline and within TBD inches (radial) of the vehicle reference centerline.

5.2.3 Gimbal Mount friction

The torque to overcome the static friction resistance of the gimbal bearing shall not exceed TBD in.-lb in a non-firing condition or TBD in.-lb in a firing condition.

5.3 FLUIDS INTERFACE

The location, quantity, and physical characteristics of fluid interface connections are specified on Figure TBD. Connections shall be capable of being connected or disconnected for purposes of engine replacement, LRU replacement, and engine checkout and maintenance. Operational characteristics and requirements, such as flowrates, flow times, cleanliness levels, are specified in section 6.0 of this document.

5.3.1 Propellant Inlets

The positions of the engine propellant inlets relative to the engine coordinate system is shown in Figure F-5.1. Inlet line diameters are shown in Table T-5.3. Propellant inlet lines are to have a minimum of TBD inches of straight duct upstream of the engine interface. Details of inlet duct interface flanges are TBD. Flexible joints in interface lines will be engine supplied.

Table T-5.3. *Propellant Inlet Line Diameters*

	Dia (in.)
Methane	11.6
Oxygen	12.2

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5.4 ELECTRICAL INTERFACE

The location, quantity, and physical characteristics of electrical interface connections are specified on drawing TBD. Connections shall be capable of being connected or disconnected for purposes of engine replacement, LRU replacement, and engine checkout and maintenance. Operational characteristics and requirements are specified in section 6.0 of this document.

5.5 DATA AND COMMAND INTERFACE

The location, quantity, and physical characteristics of data and command interface connections are specified on drawing TBD. Connections shall be capable of being connected or

disconnected for purposes of engine replacement, LRU replacement, and engine checkout and maintenance. Operational characteristics and requirements are specified in section 6.0 of this document.

6.0 OPERATIONAL CHARACTERISTICS AND REQUIREMENTS

6.1 DESCRIPTION

The Space Transportation Booster Engine (STBE) employs a bipropellant gas generator cycle using liquid methane as fuel and liquid oxygen as oxidizer. Figure F-6.1 shows a propellant flow diagram of the STBE. Two high-pressure turbopump units are employed and driven by the one gas generator. The methane pump consists of two stages while the LO_2 pump is a single-stage unit. LCH_4 is used to cool the main chamber and the tubular nozzle.

6.2 PRELAUNCH

The engine shall be able to achieve sufficient cooldown within TBD minutes from the time propellants are supplied to the engine. Maximum cooldown flows are TBD. The engine control shall provide an engine ready signal to the vehicle prior to start.

6.3 STARTING

The STBE shall have a self-contained control system to regulate the startup sequence. Time from start signal to NPL shall be less than five seconds, with a maximum thrust buildup rate TBD.

6.3.1 Engine Electrical Start Sequence Requirements

TBD

6.3.2 Engine Start Transient Characteristics

TBD

6.4 STEADY STATE

The STBE shall be designed to operate at a normal power level (NPL) of 644,898-pound sea level thrust. Table T-6.4.A shows performance parameters at NPL. Table T-6.4.B summarizes some of the major parameters from the previous table for quick reference.

6.4.1 Steady-State Performance Limits

Performance shall be delivered within the following limits:

Thrust, Sea Level — 644.9K at NPL ± 3%

Inlet Mixture Ratio — 2.7 at NPL ± 2%

6.4.2 Uncoupled Thrust

The engine-produced uncoupled oscillatory thrust shall be within the following limits:

Frequency (Hz)	*Thrust Limits (lb)*
0 — 0.5	± 6000
0.5 — 1.5	± 1500
1.5 — 25	± 450
26 — 100	± 1500

Post-transient oscillations of main chamber pressure shall be within ±5 percent of mean steady state with damp time less than 150 ms.

6.4.3 Burn Duration

Maximum burn duration shall be 160 seconds at NPL.

6.5 ENGINE SHUTDOWN

6.5.1 Shutdown Sequence

The engine shall be designed to shutdown from any power level safely. The maximum shutdown time from NPL to zero is TBD.

6.5.2 Shutdown Transient Characteristics

The maximum thrust decay rate shall not exceed TBD-pound thrust change in TBD ms. Shutdown impulse is TBD.

6.6 FLUID REQUIREMENTS

6.6.1 Fuel

Liquid methane is to be provided in accordance with TBD. However, an example of the typical composition of propellant grade methane is shown in Table T-6.6.A. Fuels are to be supplied at the flowrate, minimum pressure, and temperature outlined in Table T-6.6.B.

Fuel inlet temperature limits, pressure limits, and min NPSP limits for steady state and starting are shown by the box in Figure TBD.

6.6.2 Oxidizer

Liquid oxygen is to be supplied in accordance with specification MIL-P-25508E-3 and at the flow rate, pressure, and temperature outlined in Table T-6.6.C.

Oxidizer inlet temperature limits, pressure limits, and minimum NPSP limits for starting and steady state are shown in Figure TBD.

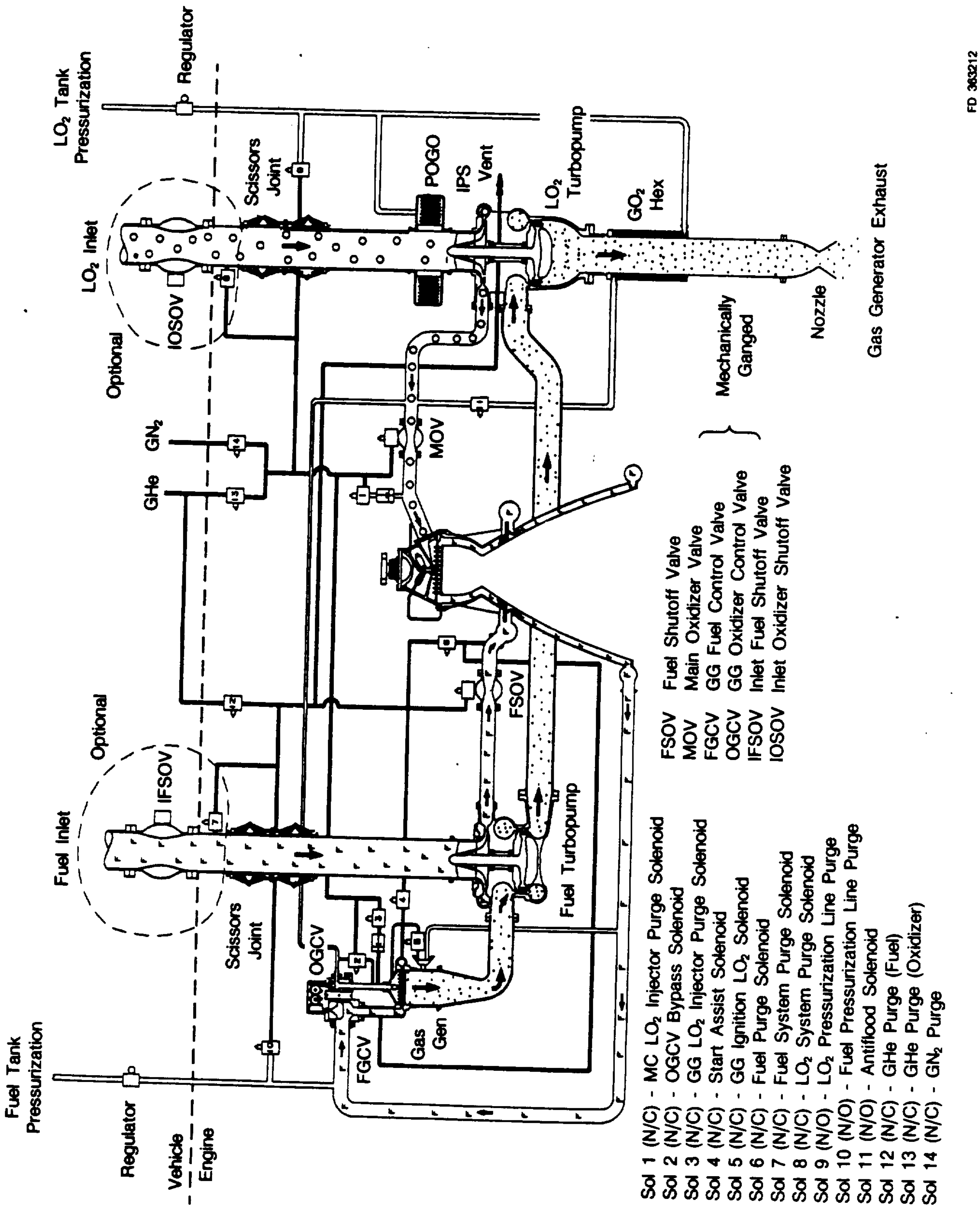
LO2 Tank Pressurization
Regulator
LO2 Inlet
Optional
IOSOV
Scissors Joint
POGO
IPS Vent
LO2 Turbopump
GO2 Hex
Gas Generator Exhaust
Nozzle
Mechanically Ganged
GN2
GHe
MOV
FSOV
Optional
IFSOV
Fuel Inlet
Fuel Turbopump
Scissors Joint
OGCV
Gas Gen
FGCV
Fuel Tank Pressurization
Regulator
Vehicle
Engine
FSOV Fuel Shutoff Valve
MOV Main Oxidizer Valve
FGCV GG Fuel Control Valve
OGCV GG Oxidizer Control Valve
IFSOV Inlet Fuel Shutoff Valve
IOSOV Inlet Oxidizer Shutoff Valve
Sol 1 (N/C) - MC LO2 Injector Purge Solenoid
Sol 2 (N/C) - OGCV Bypass Solenoid
Sol 3 (N/C) - GG LO2 Injector Purge Solenoid
Sol 4 (N/C) - Start Assist Solenoid
Sol 5 (N/C) - GG Ignition LO2 Solenoid
Sol 6 (N/C) - Fuel Purge Solenoid
Sol 7 (N/C) - Fuel System Purge Solenoid
Sol 8 (N/C) - LO2 System Purge Solenoid
Sol 9 (N/O) - LO2 Pressurization Line Purge
Sol 10 (N/O) - Fuel Pressurization Line Purge
Sol 11 (N/O) - Antiflood Solenoid
Sol 12 (N/C) - GHe Purge (Fuel)
Sol 13 (N/C) - GHe Purge (Oxidizer)
Sol 14 (N/C) - GN2 Purge
FD 363212

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Table T-6.4.A. *STBE Design at Normal Power Level*

PRATT & WHITNEY
GAS GENERATOR CYCLE OFF-DESIGN DECK
STBE ENGINE STUDY

ENGINE PERFORMANCE

VACUUM THRUST	711823.
SEA LEVEL THRUST	644898.
VACUUM IMPULSE	328.35
SEA LEVEL IMPULSE	297.48
TOTAL ENGINE INLET FLOW RATE	2176.6
OVERALL ENGINE MIXTURE RATIO	2.70

CHAMBER PERFORMANCE

PRESSURE	2250.0
TEMPERATURE	6601.7
THRUST	679922.
IMPULSE	342.90
FLOW RATE	1982.9
THROAT AREA	162.71
NOZZLE AREA RATIO	28.
MIXTURE RATIO	3.48
NOZZLE EFFICIENCY	0.965
CSTAR EFFICIENCY	0.980

ENGINE HEAT TRANSFER

CHAMBER COOLANT DP	1806.
CHAMBER COOLANT DT	177.
CHAMBER Q	66563.
NOZZLE COOLANT DP	534.
NOZZLE COOLANT DT	333.
NOZZLE Q	41724.

GAS GENERATOR PERFORMANCE

PRESSURE	1687.5
TEMPERATURE	1800.0
THRUST	31901.
IMPULSE	172.44
FLOW RATE	185.0
MIXTURE RATIO	0.301
NOZZLE EFFICIENCY	0.970
NOZZLE GAS CONSTANT	97.2
NOZZLE GAMMA	1.177
NOZZLE AREA	88.7

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	201.0	588.3	123.1	26.40
1ST STAGE EXIT	2325.1	216.0	588.3	145.4	26.55
MAIN PUMP EXIT	4621.3	230.3	588.3	167.2	26.74
FSOV INLET	4506.9	231.0	588.3	167.2	26.68
FSOV EXIT	4451.2	231.3	588.3	167.2	26.64
CHAM/COOL INLET	4369.8	231.8	442.2	167.2	26.60
CHAM/COOL EXIT	2563.5	408.8	442.2	317.8	15.79
CH INJ INLET	2542.0	408.6	442.2	317.8	15.75
NOZ/COOL INLET	4024.2	233.8	146.0	167.2	26.40
NOZ/COOL EXIT	3490.4	566.7	146.0	453.0	10.42
TANK PRESS OUT	3249.7	563.0	3.8	453.0	9.93
TANK PRESS IN	47.0	421.4	3.8	453.0	0.17
FGCV INLET	3249.7	563.0	142.3	453.0	9.93
FGCV EXIT	2385.1	545.3	142.3	453.0	7.91
GG INJ INLET	2276.1	542.4	142.3	453.0	7.63

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1588.3	61.6	71.17
MAIN PUMP EXIT	3338.3	178.8	1588.3	72.8	71.74
GOX HEX IN	3228.9	179.3	5.0	72.8	71.58
TANK PRESS IN	47.0	720.0	5.0	275.4	0.22
MOV INLET	3228.9	179.3	1540.6	72.8	71.58
MOV EXIT	2647.4	181.7	1540.6	72.8	70.70
CH INJ INLET	2552.5	182.0	1540.6	72.8	70.55
OGCV INLET	2880.3	180.7	42.7	72.8	71.05
OGCV EXIT	2668.2	181.6	42.7	72.8	70.73
GG INJ INLET	2434.7	182.5	42.7	72.8	70.37

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
FUEL TURB INLET	1532.7	1800.0	185.0
FUEL TURB EXIT	646.7	1655.5	185.0
LOX TURB INLET	563.9	1645.4	185.0
LOX TURB EXIT	289.1	1550.3	185.0
NOZZLE INLET PRES	221.6		

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Table T-6.4.A. *STBE Design at Normal Power Level (Continued)*


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************************************
*          PRATT & WHITNEY         *
* GAS GENERATOR CYCLE OFF-DESIGN DECK *
*         STBE ENGINE STUDY        *
************************************
```

TURBOMACHINERY PERFORMANCE DATA

FUEL TURBINE.

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.774	0.759
HORSEPOWER	19000.	17763.
SPEED (RPM)	10673.	10673.
S SPEED	35.2	44.3
S DIAMETER	1.84	1.54
MEAN DIAMETER (IN)	19.12	19.10
VEL.RATIO (ACTUAL)	0.47	0.48
MAX TIP SPEED	918.	941.
BLADE HEIGHT	0.58	1.10
AN SQUARED	39.7	75.2
EFFECTIVE AREA	14.07	21.17
PRES.RATIO (T/T)	1.54	1.54
GAS CONSTANT (FT)	97.20	
GAMMA	1.1626	

FUEL PUMP

	STAGE ONE	STAGE TWO
EFFICIENCY	0.713	0.729
HORSEPOWER	18570.	18192.
SPEED (RPM)	10673.	10673.
NPSH (FT)	177.7	12461.5
SS SPEED	27007.	1130.
S SPEED	910.	905.
HEAD (FT)	12373.	12405.
DIAMETER (IN)	18.28	18.28
TIP SPEED (FT/SEC)	852.	852.
VOL FLOW	10002.	9945.
HEAD COEF	0.5396	0.5410
FLOW COEF	0.0728	0.0724

LOX TURBINE

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.774	0.699
HORSEPOWER	12630.	12637.
SPEED (RPM)	7601.	7601.
S SPEED	50.9	55.4
S DIAMETER	1.16	1.06
MEAN DIAMETER (IN)	18.90	18.77
VEL.RATIO (ACTUAL)	0.40	0.40
MAX TIP SPEED	680.	715.
BLADE HEIGHT	1.60	2.77
AN SQUARED	54.9	94.4
EFFECTIVE AREA	38.67	50.39
PRES.RATIO (T/T)	1.34	1.43
GAS CONSTANT (FT)	97.05	
GAMMA	1.1697	

LOX PUMP

EFFICIENCY	0.756
HORSEPOWER	25267.
SPEED (RPM)	7601.
NPSH (FT)	62.4
SS SPEED	37052.
S SPEED	1037.
HEAD (FT)	6618.
DIAMETER (IN)	18.91
TIP SPEED (FT/SEC)	628.
VOL FLOW	10017.
HEAD COEF	0.5406
FLOW COEF	0.0821

VALVE DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL SHUT OFF VLV	55.7	22.83	588.3	1.24
FUEL GG VALVE	864.5	2.803	142.3	26.60
MAIN OXID VALVE	581.6	11.30	1540.6	18.01
LOX GG VALVE	212.0	0.521	42.7	7.36

INJECTOR DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL GG INJ	588.6	3.960	142.3	25.86
FUEL CH INJ	292.0	13.19	442.2	11.49
LOX GG INJ	747.2	0.279	42.7	30.69
LOX CH INJ	302.5	15.78	1540.6	11.85

Table T-6.4.B. *Steady-State Performance Summary*

	NPL
% Thrust	100
SL Thrust, lb	644,898
Vac Thrust, lb	711,823
Del SL I_{sp}, sec	297.5
Del Vac I_{sp}, sec	328.4
Chamber Pressure, psia	2250
Inlet O/F Ratio	2.7
Chamber O/F Ratio	3.48
LCH_4 Flowrate, lb/sec	588.3
LO_2 Flowrate, lb/sec	1588.3

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Table T-6.6.A. *Sample Composition of Propellant Grade Methane*

*Propellant Grade Methane Specification No. SG-20A**

Product	*Methane, CH_4*
Physical State	*Gas or Liquid*
Purity	*Minimum 99.9%*
Typical Impurities	
Oxygen	10 ppm
Nitrogen	100 ppm
Ethane	200 ppm
Ethylene	200 ppm
Propane	100 ppm
Carbon Monoxide	20 ppm
Carbon Dioxide	10 ppm
Hydrogen Sulfide	<1 ppm
Water	<10 ppm
Quantities Available	
Tube Trailers	up to 200,000 SCF at 2,400 psig
Cryogenic Liquid Trailers	up to 9,500 gallons

*Supplied by Liquid Carbonic, Specialty Gas Corp., Chicago, IL

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Table T-6.6.B. *Fuel Inlet Conditions at Steady State*

	NPL
LCH_4 Temperature, °R	201.0
LCH_4 Pressure, psia	47.0
LCH_4 Flowrate, lb/sec	588.3

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Table T-6.6.C. *LO_2 Inlet Conditions at Steady State*

	NPL
Temperature, °R	164.0
Pressure, psia	47.0
Flowrate, lb/sec	1588.3

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6.6.3 Propellant Vent Requirements

TBD

6.6.4 Pneumatic Requirements

TBD

6.6.5 Hydraulic Requirements

TBD

6.6.6 Tank Pressurization Gas

Tank pressurization gas will be extracted from the engine and supplied at the interface connections at the conditions provided in Table T-6.6.D.

Table T-6.6.D. *Tank Pressurization Gas*

	Temperature (°R)	Pressure (psia)	Flow (lb/sec)
Oxygen	TBD	1000	TBD
Methane	TBD	1000	TBD

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6.7 ELECTRICAL REQUIREMENTS

The Space Transportation Booster Engine (STBE) requires electrical interfaces for the purposes powering the engine controller and control valve solenoids, transmission of vehicle generated control signals, and data transmission.

6.7.1 Electrical Power

Electrical power shall be supplied in accordance with specification MIL-STD-704D. Connections shall be capable of being connected or disconnected without damage for purposes of routine maintenance, engine replacement, or replacement of line replaceable units (LRU's). Connections are to remain secure during flight.

6.7.2 Overall Power Requirements

The STBE has the following electrical power requirements:

DC	:	TBD volts
		TBD amps
		TBD watts

Allowable voltage variations for dc currents are TBD.

6.7.3 Transient Voltage Limits

Normal transient voltage shall not exceed TBD volts for TBD msec. The engine controller shall withstand normal voltage transients without adverse effects or going into a hold mode.

7.0 VEHICLE/ENGINE DATA AND COMMUNICATION INTERFACE

The vehicle will contain all the logic necessary to generate engine control system commands and to receive and store engine generated status, faults, and performance data. Vehicle generated commands will be either discrete or variable. discrete commands will be issued to begin the start sequence, shutdown sequence, etc. Variable commands will consist of thrust level commands and mixture ratio commands. Communication between the vehicle and engine control will be through a TBD databus. The communication system will be capable of transmitting TBD words at a TBD data rate.

8.0 ENVIRONMENTAL CRITERIA

8.1 PRESSURE ENVIRONMENT

TBD

8.2 TEMPERATURE ENVIRONMENT

TBD

8.2.1 Ambient Temperature Limits

TBD

8.2.2 Engine Surface Temperature

All engine surfaces will be maintained at sufficient temperature to prohibit the formation of liquid air.

8.3 ACOUSTIC ENVIRONMENT

The Space Transportation Booster Engine (STBE) will withstand acoustic impingement loads of TBD db sound pressure level over the frequency range of TBD Hz to TBD Hz.

8.4 VEHICLE BASE HEATING

The STBE will not impose thermal loads on the base of the vehicle in excess of the following:

Max Heat Flux	:	TBD
Max Surface Temperature	:	TBD
Max Heat Transfer Coefficient	:	TBD

9.0 ENGINE LOADS

9.1 ALLOWABLE GIMBAL LOADS

The Space Transportation Booster Engine (STBE) can be gimbaled through an angle of ±6 degrees about the engine centerline in a square pattern while firing. The estimated maximum allowable gimbal actuation loads are shown in Table T-9.1.

Table T-9.1. Estimated STBE Maximum Gimbal Actuation Loads

Max Actuator Load, lb	TBD
Max Gimbal Rate, degrees/sec	10
Max Gimbal Acceleration, rad/sec	10

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9.2 ACCELERATION LOADS

9.2.1 Vibration Loads

The engine will withstand self-induced vibrations without harmful effects. All subsystems must be designed so that no combination of loads (engine operation forces, actuator induced loads, aerodynamic loads, vibroacoustic induced loads) will render detrimental engine dynamics.

9.2.2 Liftoff Loads Through Landing

The engine design shall be able to sustain simultaneous occurrence of the following low frequency quasistatic, c.g. acceleration for launch, ascent, descent, and landing:

	Load (g's)
Axial	± 6
Lateral (y)	± 2
Lateral (z)	± 4

9.3 GROUND HANDLING LOADS

The STBE shall be capable of withstanding TBD-pound ground handling loads applied in any direction while installed in a handling frame. The STBE will also be capable of withstanding TBD-pound axial and TBD-pound lateral applied in any combination while supported on the vehicle by the normal connecting interfaces.

10.0 GROUND SUPPORT AND MAINTENANCE

The design of the STBE will optimize maintainability and serviceability. The capability of on-pad/vehicle servicing will be maximized over field level (off-vehicle) service, which in turn will be preferred over factory service.

10.1 PRELAUNCH GROUND SERVICE

There will be no requirement for ground service equipment within 24 hours after propellants are loaded in vehicle. The engine will be capable of achieving thermal conditioning without ground service equipment.

10.2 LEAK DETECTION

The capability to detect helium leakage at separable connections will be provided. The helium leakage flowrate is not to exceed 0.0001 scc/sec at leak check pressure.

10.3 MAINTENANCE AND SERVICE INTERVALS

10.3.1 Inspection Interval

TBD

10.3.2 Line Replaceable Unit (LRU) Replacement Interval

TBD

10.3.3 Engine Replacement Interval

TBD

10.4 ENGINE MAINTENANCE ACCESS

Sufficient access to the engine shall be provided to carry on engine maintenance operations per Sections 10.4.1 through 10.4.3.

10.4.1 Routine Maintenance Access

Access will be provided to allow complete external visual inspection 360 degrees around the engine powerhead. In addition, sufficient access will be provided to allow periodic internal inspection including the installation of internal inspection equipment and rotation of the turbopumps. Access requirements also include the ability to perform periodic leak checks and functional checks of all powerhead systems.

10.4.2 Corrective Maintenance Access

Space will be provided, when the engine is installed in the vehicle, to replace all LRU's (TBD).

10.4.3 Engine Positioning

The capability of positioning and locking the engine within the gimbaling envelope will be provided.

4.1.1.7.9 *Preliminary Contract End Item Specification*

The following Preliminary Contract End Item Specification for the STBE is submitted in its entirety with its unique paragraph, figure, table, and page numbering.

PRELIMINARY
SPACE TRANSPORTATION BOOSTER ENGINE
CONTRACT END ITEM
SPECIFICATION

CONCURRENCE:

P&W — ENGINEERING MANAGER

P&W — PROJECT MANAGER

APPROVAL:

MSFC — STBE PROJECT MANAGER

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1. SCOPE

1.1 Scope. This document establishes the requirements and characteristics of the Space Transportation Booster Engine (STBE) for use in the Advanced Space Transportation launch system.

1.2 Classification. The STBE is a regeneratively cooled, turbopump-fed bipropellant gas generator engine with a single chamber and a rated thrust of 644,898 pounds at sea level conditions. Propellants are liquid oxygen and liquid methane used at a rated inlet oxidizer-to-fuel mixture ratio of 2.7:1. Rated engine thrust is achieved at an estimated design chamber pressure of 2250 psia with an overall nozzle area ratio of 28:1. The engine has a rated specific impulse of 297.5 seconds at sea level conditions. The operating life of the engine is an accumulated running time of 16.67 hours and/or 100 starts.

2. APPLICABLE DOCUMENTS

2.1 Government Documents. The following documents forms a part of this specification to the extent specified herein. If the issue of a document is not specified, the issue in effect on the baseline (Date TBD) shall be applicable. The subtier documents referenced within these documents are also a part of this specification and are applicable to the extent required by the documents specified below. The specific issue of second-tier documents referenced within these first-tier documents shall not be restricted to the baseline date specified above, and later issues of second-tier documents acceptable to the contractor may be used provided that the use of later issues does not affect the Class I change definition for the end item as defined by Configuration Management Program Plan, (Spec No. TDB). Later issues of second-tier documents unacceptable to the contractor or impacting the Class I definition above shall be forwarded to the Procuring Agency by Engineering Change Proposal (ECP). The specific issue of subsidiary documents below the second-tier level shall not be subject to configuration management controls.

SPECIFICATIONS

Federal

MSFC-SPEC-234A	Nitrogen, Space Vehicle Grade, 27 July 1967
TBD	Methane, Liquid

Military

MIL-B-5087B(2)	Bonding, Electrical, Lighting, Protection for Aerospace Systems
DOD-D-1000B(3)	Drawings, Engineering and Associated Lists
MIL-I-6181D Notice 6	Interference Control Requirements, Aircraft Equipment
MIL-P-25508E(3)	Propellant, Oxygen — Type II, Grade A or Equivalent
MIL-P-27401C 20 January 1975	Propellant Pressurizing Agent, Nitrogen

MIL-P-27407A	Helium, Type I, Grade A (Gaseous) or Equivalent
TBD	Propellant, Methane
MIL-S-7742B Notice 2	Screw Threads, Standard, Optimum Selected Series, General Specifications for
MIL-S-8879A Notice 2	Screw Threads, Controlled Radius Root With Increased Minor Diameter, General Specifications for

STANDARDS

Military

MIL-STD-130F	Identification Marking of U.S. Military Property
MIL-STD-704D	Electric Power, Aircraft, Characteristics and Utilization of

2.2 Other Publications. The following documents of the exact issue shown, form part of this specification to extend specified herein.

Aerospace Material Specification

AMS 2630A-80	Ultrasonic Inspection
AMS 2635C-81	Radiographic Inspection
AMS 2640J-83	Magnetic Particle Inspection
AMS 2645H-83	Flourescent Penetrant Inspection
AMS 3159C-67	Leak Test Solution Liquid Oxygen Compatible

Technical society and technical specifications and standards are generally available for reference from libraries. These documents are also distributed among technical groups and using Federal agencies.

2.3 Use of Applicable Documents. The use of publications which by reference are made supplementary or part of those listed herein shall not be mandatory. In the event of conflict between documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

3.0 REQUIREMENTS

3.1 Definitions

3.1.1 General Description. The STBE is a fixed thrust engine, capable of sea level start and continuous operation from sea level to altitude. The engine power cycle is a regeneratively-cooled gas generator cycle using the liquid methane fuel as a coolant for the main combustion chamber of nozzle. The engine shall be capable of operating at a rated thrust level of 490,604 pounds at

vacuum conditions. Propellants are liquid methane and liquid oxygen. The engine is capable of being gimbellied ±6° square pattern in the vehicle pitch and yaw planes. The engine is capable of being used in either single of multi-engine vehicle stages, and may be used in either an expendable or reusable vehicle configuration.

3.1.2 Missions. TBD

3.1.3 Operational Concepts. TBD

3.1.4 Organizational and Management Relationships. TBD

3.1.5 Systems Engineering Requirements. TBD

3.1.6 Government Furnished Property List. TBD

3.2 CHARACTERISTICS

3.2.1 Performance

3.2.1.1 Primary Performance Characteristics.

3.2.1.1.1 Ratings. The performance ratings and tolerances shall be as listed in Table I.

Table I. *STBE Rocket Engine Performance Ratings (Performance Ratings of the STBE Rocket Engine at Sea-Level Conditions, Nominal Power Level and 2.7 Mixture Ratio)*

Parameter	*Ratings and Tolerances* *Steady State*
Engine Thrust (lb_f) — Sea Level	644,898 ± 19,347
Specific Impulse ($lb_f/lb_f/sec$)	297.5 ± 1.5

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3.2.1.1.2 Duty cycle. The limiting duty cycle for the engine consists of one start during a single mission with a running duration of 160 seconds.

3.2.1.2 Secondary Performance Characteristics

3.2.1.2.1 Engine Life. After completion of the quality conformance test, the engine shall achieve the primary performance ratings and tolerances in the specific environments for the following service life with a minimum use of maintenance, adjustment, and servicing. The engine shall have an operating life of at least 16.67 hours/100 firings during which the performance specified in 3.2 shall be achieved. Nonfiring functional checks of the complete engine systems shall not exceed 500 cycles. The engine shall be capable of completing this operating life after accumulating a storage life of TBD years under the applicable environments. The engine shall be designed to have a total life of TBD years. The engine can be overhauled and the operating life returned to the maximum during this period.

3.2.1.2.2 Limited Life Parts Replacement. The engine has no limited life parts for operation within the overhaul life (25 missions).

3.2.1.3 Operating Essentials. The rocket engine shall start, operate, and shutdown in accordance with the performance of 3.2 and the following operating essentials throughout the applicable environmental conditions. The engine shall be capable of autonomous control during all phases of operation from prestart through post shutdown. The engine shall require an external start and shutdown signal.

3.2.1.3.1 Prelaunch. There is no requirement for service from ground equipment within 24 hours after propellants are loaded. The turbopump will achieve thermal conditions without ground servicing in less than TBD minutes from the time propellants are supplied to engine with maximum thermal conditioning flowrates TBD. Propellant leakage, either external or internal, shall not occur in such a manner as to impair or endanger engine/vehicle function. Leakage isolation capability shall be provided with design objective that separable connections not exceed 1×10^{-4} scc/sec helium at leak check pressure.

3.2.1.3.2 Start. The starting system shall be such that the engine can be safely started on the first attempt. At the initiation of the start cycle, propellant inlet conditions must be maintained within the limits defined in Figures SP-1, SP-2, and SP-3. During the start cycle, engine control shall be self-contained. The engine will start and accelerate to normal power level (NPL) in less than five seconds. The maximum thrust buildup shall not exceed TBD pounds in any TBD msec time interval. Expendable igniters and hypergolic fluids may be used. Nozzel prefill shall not be allowed.

3.2.1.3.3 Steady State. During steady-state operation, the engine thrust, chamber mixture ratio, and gas generator mixture ratio will be controlled using the gas generator fuel valve, main oxidizer valve, and gas generator oxidizer valve, respectively. The control of thrust shall be such that the rocket engine operates within the limits specified in 3.2.1.1.1. Thrust oscillations about the engine steady-state operating thrust level shall not exceed the following:

Range	*Variations*
0 — 0.5 Hz	$< \pm$ 6000 lb
0.5 — 1.5 Hz	$< \pm$ 1500 lb
1.5 — 25 Hz	$< \pm$ 450 lb
25 — 100 Hz	$< \pm$ 1500 lb

Thrust during the starting transient shall not exceed 656,250 pounds. Mixture ratio variation during steady state shall not exceed $\pm$ 1.0 percent of nominal value.

3.2.1.3.4 Shutdown. Provisions incorporated for power cutoff shall be such that a positive and safe shutdown can be achieved under all engine normal operating conditions. The deceleration time from NPL to zero thrust shall not exceed TBD seconds. The thrust decay shall not exceed TBD pounds in any TBD msec internal. The shutdown impulse from NPL to TBD lb^f -sec.

3.2.2 Physical Characteristics

3.2.2.1 Dimensions. The engine dimensions and interface characteristics shall be as provided on the installation drawing included in the ICD.

3.2.2.2 Rocket Engine Dry Weight. The dry weight of the engine shall not exceed 6960 pounds. The weight of the propellants in the engine at normal operating conditions shall not exceed 809 pounds. The burnout weight shall be the same as engine dry weight. There shall be no liquid propellants remaining in the engine after normal venting.

3.2.3 Reliability. The engine shall be designed for a minimum reliability of 0.99 at 90 percent confidence for operation at the normal mission power level and burn duration and in the nominal mission environment. The probability of a failure that can result in damaging or destruction release of energy while operating under conditions of a nominal mission shall be no greater than 1-0.999.

3.2.4 Maintainability. Design features shall be incorporated to assure effective engine maintenance, refurbishment, and repair. Maintenance requirements shall be minimized. The engine shall be capable of service and maintenance in either the vertical or horizontal position.

3.2.4.1 Engine removal and replacement. The engine shall be capable of removal and replacement when installed in the vehicle oriented in either the vertical or horizontal position. This activity shall be accomplished by a crew of TDB men (maximum) in TBD hours (maximum).

3.2.4.2 Line Replacement Units (LRU). All LRUs shall be interchangeable from engine to engine and shall be inspectable while mounted on the engine installed in a vehicle in the vertical or horizontal position. LRUs shall be removable and replaceable within the crew and time limits specification Table TBD.

3.2.4.2.1 Configuration Item. All configuration items shall be interchangeable from engine to engine and shall be inspectable while mounted on the engine installed in a vehicle in the vertical or horizontal position. Configuration items shall be removable and replaceable within the crew and time limits specified in table TBD.

3.2.4.3 Inspectability. All critical engine components, including bearing seals, filters, and structural welds shall be inspectable in the assembled engine.

3.2.4.4 Maintenance Equipment. Special maintenance equipment and support equipment shall be minimized.

3.2.5 Operational Availability. TBD

3.2.6 STBE Safety. Safety shall be in accordance with NHB 1700.1 (IV-a), V3 and NHB 5300.4 (ID-2).

3.2.6.1 Safety Design Preferences. The STBE shall, in the following order of preference, be designed to eliminate hazards by appropriate design measures; or prevent hazards through use of safety devices or features; or control hazards through use of warning devices, special procedures, and emergency protection subsystems.

3.2.6.2 Materials. STBE materials shall be selected with characteristics which do not present hazards to personnel or equipment in their intended use or environment.

3.2.6.3 Isolation of Hazardous Conditions. Provisions shall be made to physically isolate or separate hazardous, incompatible subsystems, materials, or environments.

3.2.6.4 Purging, Venting, Drainage, Detection. Provisions shall be made to prevent hazardous accumulations of gases or liquids (i.e., toxic explosive, flammable or corrosive).

3.2.6.5 Drain, Vent and Exhaust Port Design. Drains, vents, and exhaust ports shall prevent exhaust fluids, gases, or flames from creating hazards to personnel, vehicle, or equipment.

3.2.6.6 Protection of Critical Functions. Subsystems shall be designed to prevent inadvertent or accidental activation or deactivation or safety-critical functions or equipment, which would be hazardous to personnel or vehicles during flight and ground operations.

3.2.6.7 Pressure Vessel Protection. Pressure vessels shall be protected against overpressurization or underpressurization which could be hazardous to personnel or hardware.

3.2.7 Environment. The engine shall be capable to accomplishing the intended functions under environmental conditions imposed upon the engine during the engine service life. The engine shall be capable of operating in a single or multi-engine installation in any static firing environment equivalent to prelaunch and flight conditions defined in this document.

The engine during its service life shall not suffer any detrimental effects during or after exposure to the following environmental conditions.

3.2.7.1 Natural Environment. TBD

3.2.7.2 Induced Environment

3.2.7.2.1 Vibration. The engine shall withstand any self-induced vibrations without deleterious effect on the engine or impairment of its serviceability. The engine shall also withstand externally imposed vibration environmental conditions from any source to the extent shown below:

Direction	*Level*
Axial	TBD g's
Lateral	TBD g's

3.2.7.2.2 Acoustic Environment. The engine shall withstand acoustic impingement on TBD db overall sound pressure level over the frequency range of TBD Hz for TBD minutes.

3.2.7.2.3 Engine Side Loads. The engine shall be designed to withstand its self-induced side loads caused by nozzle flow separation during on-pad thrust buildup and decay. The peak transient force shall be no greater than specified in para. 3.2.1.3.2.

3.2.7.2.4 Flight Loads. The maximum allowable flight maneuver loads are:

Direction	*Level*
Axial	TBD pounds
Lateral	TBD pounds

The engine shall withstand during flight, without permanent deformation or failure, the maximum force resulting from all critical combinations of these load factors with those specified in 3.2.7.2.1.

3.2.7.2.5 Ground Handling Loads. The engine shall be designed to withstand TBD "g" handling loads applied in any direction while installed in a handling frame. The engine shall be designed to withstand TBD "g" axial acceleration in combination with a TBD "g" lateral acceleration during ground handling without the handling frame installed, but with the engine supported at normal interfaces as defined on the Interface Control Drawing. The maximum handling load at a single gimbal actuator attach point shall not exceed TBD "g".

3.2.8 Transportability/Transportation. TBD

3.2.9 Storage

3.2.9.1 Temperature Range. The engine, when stored in accordance with TBD shall not suffer any detrimental effects between temperatures of TBD to TBD °F.

3.3 DESIGN AND CONSTRUCTION STANDARDS

3.3.1 Selection of Specifications and Standards. All materials, parts, and processes shall be defined by standards and specifications, selected from those of Government, industry, and contractor. Rationale for the selection of contractor specifications and standards over existing higher order or precedence standards and specifications shall be compiled and maintained for historical record and shall be made available to the procuring activity upon request. This rationale shall include an identification of each higher order or precedence specification or standard examined and state why each was unacceptable. For purposes of this order or precedence, commercial materials, parts, and processes shall be considered equivalent to contractor standards.

3.3.2 General

3.3.2.1 Structural Conditions. The engine shall withstand, without impairing satisfactory operations, the maximum forces resulting from all critical combinations of the operating, interface, and environmental loading conditions specified in this specification. As a guide for design purposes, the following criteria should be used. The allowable loads and moments at the interfaces shall be specified on the Interface Control Drawing.

Structural Factors of Safety:

- Minimum Yield: (1.1 at NPL)
- Minimum Ultimate: (1.4 at NPL, combined loads)
- Minimum Ultimate: (1.5 at NPL, pressure only)
- Minimum Proof: (1.2 at NPL, unless fracture mechanics requires a higher factor)
- LCF: (4.0 at NPL)
- HCF: Design goal of infinite life, otherwise: (4.0 at NPL for A-basis materials; 10.0 otherwise)
- Creep (4.0 at NPL).

3.3.2.2 Fracture Life Verification. Turbopump components shall be analyzed in accordance with the Fracture Control Plan as specified. Components shall be designed for 1.25 on endurance limit when feasible.

3.3.3 Standards. MS, AN, or MIL standard parts shall be used wherever they are suitable for the purpose, and shall be identified by their standard part numbers. The use of nonstandard parts will be acceptable only when standard parts have been determined to be unsuitable. MS, AN, and AS design standards shall be used wherever applicable.

3.3.4 Threads. Conventional straight screw threads shall conform to the requirements of MIL-S-7742 or MIL-S-8879. Duplicate parts, differing only in thread form, shall not be permitted. Unless otherwise specified, threaded parts smaller than 0.164-inch diameter shall have threads in accordance with MIL-S-7742. When an allowance is required for applications in elevated temperature, corrosive atmosphere, or other conditions which may cause thread seizure, this allowance shall be obtained by increasing the diameters of the internal threads.

3.3.5 through 3.3.7. TBD

3.3.8 Materials. The engine structure shall be designed employing material properties based on MIL-HDBK-5E. Material property data from other sources shall state the statistical quality of the data, including methods used to derive allowables, and shall require procuring agency approval. Materials lacking a sufficient data base in the operating environment shall be tested.

3.3.8.1 LO_x/GO_2 Compatibility. Materials exposed to liquid or gaseous oxygen must meet the requirements of NHB 8060-1. This includes impact, promoted combustion, and frictional heating tests.

3.3.9 Contamination Control. TBD

3.3.10 Coordinate Systems. TBD

3.3.11 Interchangeability and Replaceability. TBD

3.3.12 Identification and Marking. TBD

3.3.13 Workmanship. TBD

3.3.14 Human Performance/Human Engineering. TBD

3.4 LOGISTICS. TBD

3.5 PERSONNEL AND TRAINING. TBD

3.6 INTERFACE REQUIREMENTS

3.6.1 Inter-Program Interface Requirements

3.6.1.1 Engine/Vehicle Interfaces. Engine/vehicle physical and functional interfaces are specified in Interface Control Documents, document numbers TBD. These interfaces include engine envelope (static and dynamic) dimensions, engine mass properties, engine/vehicle connections for mechanical attachment, for propellants and other fluids transfer, for electrical power, and for engine/vehicle communication. Also included are functional criteria such as prestart, start, run, and shutdown requirements; environmental criteria; loads criteria; and inspection and maintenance criteria.

4.0 VERIFICATION. TBD

5.0 PREPARATION FOR DELIVERY

5.1 PREPARATION FOR STORAGE AND SHIPMENT.

The contractor shall furnish a packing list with each engineer. All parts, accessories, components, and tools that are not installed on the engine, but are shipped with the engine, shall be included on the packing list. The rocket engine, components, and accessories shall be prepared for storage and shipment in accordance with the following:

a. The shipping container will provide adequate physical protection for the engine during shipment and storage.

b. The shipping container design will be sized to accommodate road and air shipment.

c. The shipping container will be designed to allow attachment of an indicator to record loads encountered during shipment.

d. The engine shall be capable of being maintained during shipment and storage in accordance with the procedures established by the Service Manual.

e. Packaging equipment will be provided to protect the engine against environmental conditions during shipment and when the engine is mounted on the vehicle stage.

6.0 NOTES. TBD

7.0 OPTIONAL FEATURES. TBD

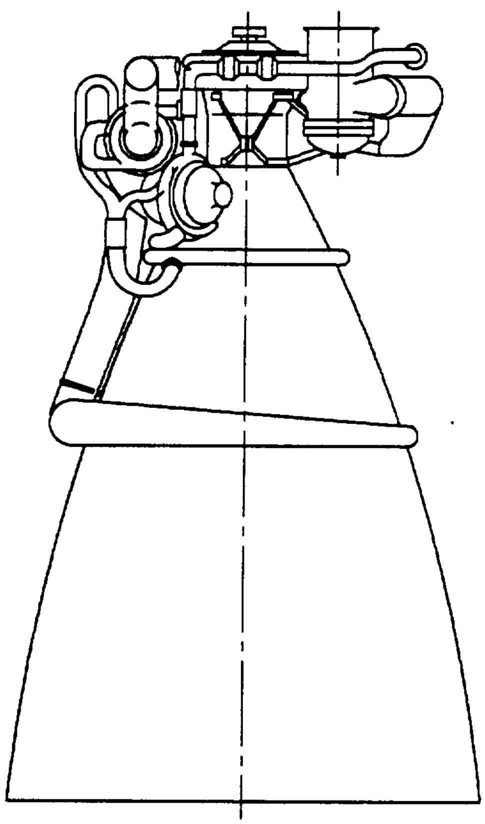
8.0 TRADE DATA. TBD

10.0 APPENDIX. TBD

4.1.2 Unique STBE Gas Generator Cycle Engine

4.1.2.1 Unique Gas Generator Engine Design Evolution

The unique LO_2/methane gas generator engine cycle study was initiated in the first quarter of 1988. The first engine design is shown in Figure 4.1.2.1-1 with engine characteristics. This engine was a 625,000-pound (625K) sea level fixed thrust with the design point at 688K sea level thrust. The first bipropellant unique engine design incorporated all of the STME/STBE low cost design and manufacturing concepts. These concepts are listed in Table 4.1.2.1-1. This was the prime expendable concept when the tripropellant was the prime reusable concept. Reliability predictions, unit production costs, and the impact on life cycle cost were evaluated for the bipropellant, expendable 625K fixed thrust sea level engine design during the first quarter of 1988. The results of these evaluations are presented in P&W Interim Report FR-19691-3.



Gas Generator Cycle

Propellants	LO_2/CH_4
Mixture Ratio	3.04
Chamber Pressure	2044 psia
Thrust - Vacuum	717,500 lb
- Sea Level	625,000 lb
Specific Impulse - Vacuum	340.1 sec
- Sea Level	296.2 sec
Nozzle Area Ratio	35
Length	145 in.
Diameter	90 in.
Weight	7014 lb

FD 359995

Figure 4.1.2.1-1. STBE Unique Gas Generator Cycle Engine — 625K Sea Level Thrust

During the second quarter of 1988, the LO_2/methane bipropellant engine concept was refined to include growth capability to 750K sea level thrust with some hardware changes. This engine design and its major characteristics are shown in Figure 4.1.2.1-2. Several design and analytical trade studies were conducted to substantiate the engine design. The major studies conducted were a boost pump trade study and a mixture ratio trade study. A summary of the boost pump and mixture ratio trade studies is included in this section.

Table 4.1.2.1-1. Design Changes To Reduce Fabrication Costs

- Simple Axial Inlet Turbopumps
- Removed MCC Igniter From Acoustic Liner for Simplification of Chamber
- Simplified MCC Coolant Channel Geometry
- Eliminated Expensive/Complex Wrap-Around Flex Lines
- Cast Turbopump Housings
- Changed to Lower Cost Materials Wherever Possible
- Equiaxed Turbine Blades
- Cast Oxygen and Fuel Pump Impellers
- Cast Gas Generator and MCC Injector Elements and Divider Plate
- Cast Chamber With Electroplate Nickel Closeout and Bicast Structural Jacket
- Filament Wound Shell on Tubular Nozzle
- Formed Tubular Nozzle

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As the bipropellant common engine study began to emerge as the focus of STBE efforts, the engine design did not undergo further study until the fourth quarter of 1988 and continued through the first quarter of 1989. This engine assembly design and overall characteristics are presented in Figure 4.1.2.1-3. This 750K engine incorporates all of the low-cost concepts as previously discussed except that the turbopumps are mounted vertically. The following paragraphs refer to the design definition of this 750K sea level thrust engine, shown in Figure 4.1.2.1-3, with low cost design and manufacturing features and vertical turbopumps.

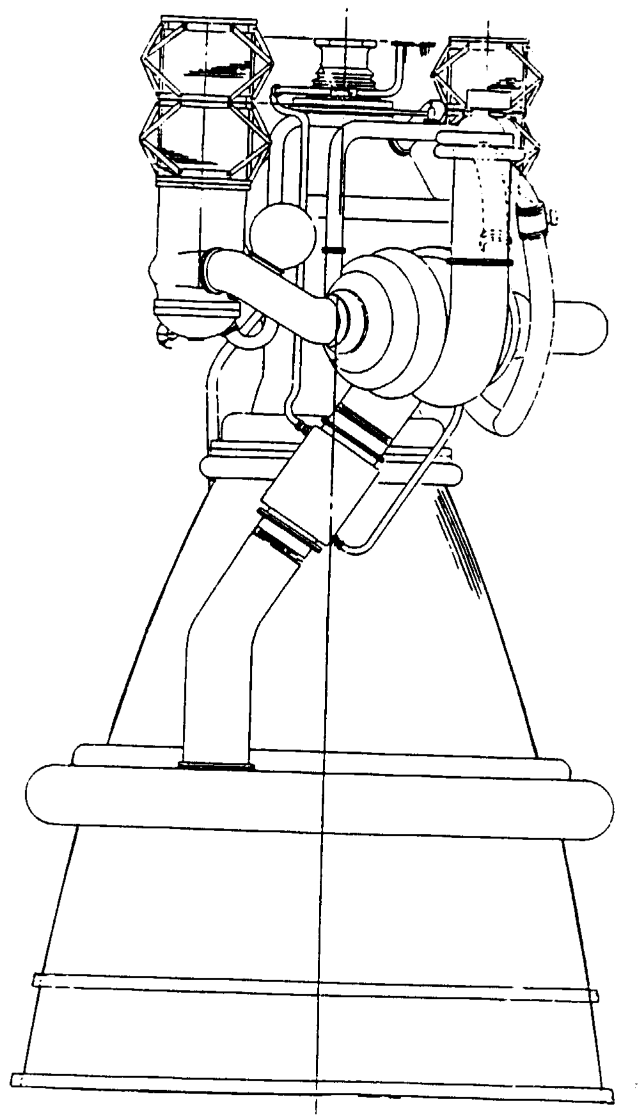
4.1.2.2 Engine Cycle

The candidate unique LO_2/CH_4 STBE configuration studied during the Phase A' extension is a gas generator cycle with liquid oxygen and liquid methane as propellants. This engine operates at a main chamber pressure of 2396 psia at the design power level (DPL) of 750,000 pounds thrust and has the capability of running at a nominal power level (NPL) of 625,000 pounds thrust. The engine has a fixed nozzle with an area ratio of 35:1 and delivers 305 seconds of sea level specific impulse at DPL. Figure 4.1.2.1-3 presents selected engine characteristics at the rated power level.

4.1.2.2.1 Flowpath Description

A simplified flow schematic, showing the major flowpaths and components for the STBE, is presented in Figure 4.1.2.2-1.

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure methane pump; thus boost pumps are not required for this system.

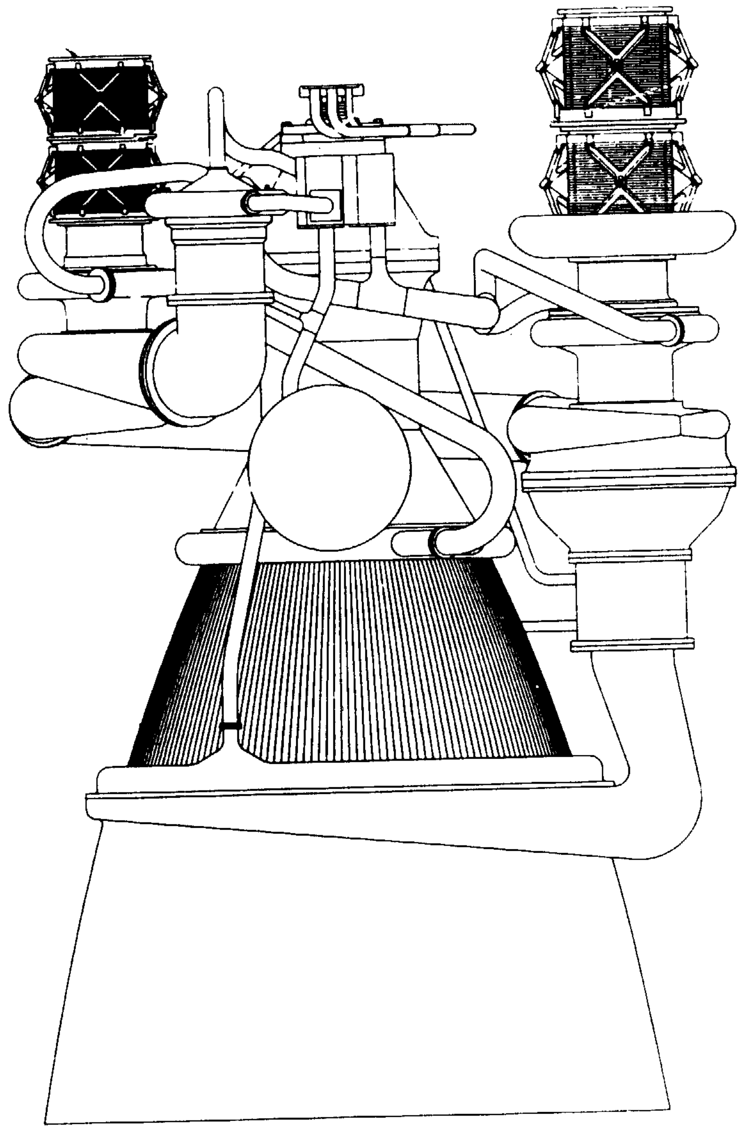


Gas Generator Cycle

Propellants	LO_2/CH_4
Mixture Ratio	3.0
Chamber Pressure	2250 psia
Thrust - Vacuum - Sea Level	709,300 lb 625,000 lb
Specific Impulse - Vacuum - Sea Level	339.0 sec 298.7 sec
Nozzle Area Ratio	35
Diameter	92 in.
Length	161 in.
Weight	6476 lb

FD 359996

Figure 4.1.2.1-2. STBE Unique Gas Generator Cycle Engine — 750K Sea Level Thrust — Growth Capability



Performance

Thrust - Vacuum	841,390 lb
- Sea Level	750,000 lb
Chamber Pressure	2396 psia
Mixture Ratio	3.0
Specific Impulse - Vacuum	342 sec
- Sea Level	305 sec
Weight	XX lb
Thrust-To-Weight	XX
Area Ratio	35
Length	89 in.
Diameter	142 in.

FD 359977

Figure 4.1.2.1-3. STBE Unique Gas Generator Characteristics at Design Power Level

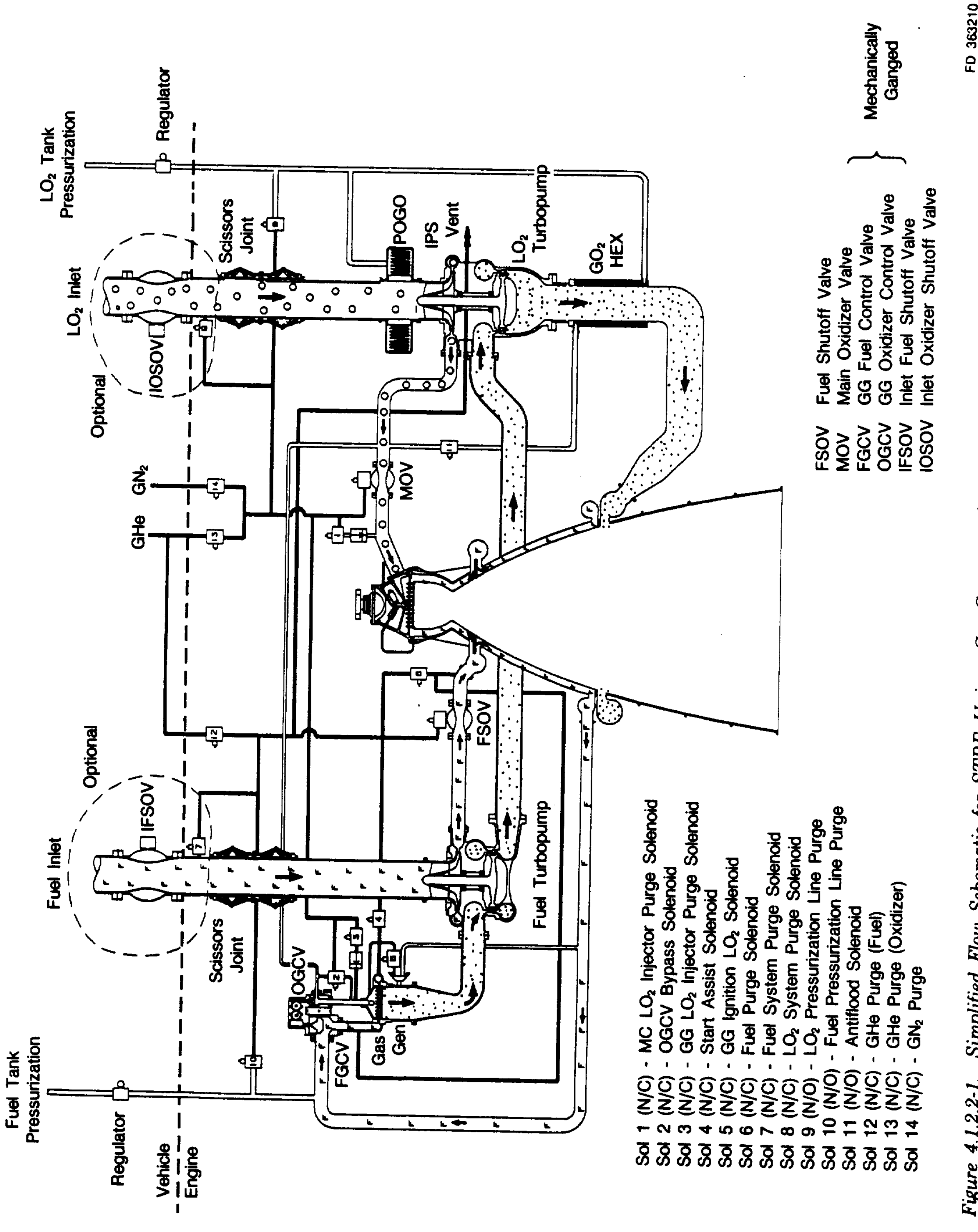


Figure 4.1.2.2-1. Simplified Flow Schematic for STBE Unique Gas Generator Cycle Engine

At the design power level, the methane pump operates at 17,181 rpm to provide the methane pressure level of 5195 psia required by the cycle. From the pump exit, the methane flows through the fuel shutoff valve to a split manifold at the inlet of the coolant passages. From the split manifold, 81.2 percent of the methane is used to regeneratively cool the milled channel, copper alloy main chamber from an area ratio of 5.48:1 back to the injector face. The remaining methane flow is used to cool the tubular stainless steel nozzle from an area ratio of 5.48:1 down to an area ratio of 35:1. This methane then flows through the fuel gas generator control valve and is injected into the gas generator to combust with some of the oxygen to provide power for the high pressure turbomachinery.

The high-pressure oxidizer pump operates at 6,787 rpm to provide the oxygen pressure level of 3046 psia required by the cycle at the design power level. From the pump exit, approximately 98.3 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

The high-pressure, high-temperature (2281 psia/1800°R at DPL) gas of the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas is initially expanded through the methane turbine and is subsequently routed to a second turbine which powers the oxygen pump. From the oxidizer turbine discharge, the flow enters a heat exchanger where energy is extracted to vaporize the oxygen being provided for tank pressurization. The turbine exhaust gas is then expanded through an area ratio of 5:1 to atmospheric pressure, providing additional thrust to the overall engine output.

4.1.2.2.2 Engine Operation

The engine will be preconditioned using liquid flow from the tanks to soak the turbopumps until they are sufficiently cooled. The inlet valves will be opened, allowing liquid from the tanks to flow down to the turbopumps and letting any vapors to percolate back up to the tank to be vented.

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor during the oxygen phase transition from gas to liquid. The transition occurs prior to fuel injection and the fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one. Pratt & Whitney has had extensive experience with oxidizer leads with the RL10 and XLR-129 engines.

With the oxidizer lead sequence, the gas generator LO_2 injector is primed prior to opening the fuel shutoff valve to ensure liquid oxygen flow, eliminating turbine temperature spikes due to oxygen phase change. A helium spin assist is also utilized to initiate turbopump rotation before the fuel is introduced into the gas generator. During the start and shutdown, a small helium purge is used in the gas generator injector and main chamber injector to eliminate the danger of hot gas flow reversals during transient operation. Gas generator and main chamber ignition will be accomplished with dual electrical spark excited, oxygen/methane torch igniters.

Main stage engine operation is open-loop controlled. The fuel gas generator control valve (FGCV), the oxygen gas generator control valve (OGCV), and the main oxidizer valve (MOV), shown in Figure 4.1.2.2-1, are used to set the engine thrust and mixture ratio. Thrust and main chamber mixture ratio are set on the ground by trimming the MOV and OGCV respectively. The gas generator mixture ratio is set using the FGCV. All valves are operated by hydraulic actuators.

Engine acceleration is accomplished by a time-based scheduling of the valves to the commanded starting level (~20 percent power level). The acceleration to full thrust is also

accomplished with open-loop valve schedules. Engine shutdown is accomplished using a time-based scheduling of the propellant valves. The OGCV is closed first to power down the turbopumps, then the MOV closes, followed by shutting off the methane system.

In addition to a normal operational mode, the engine system is capable of shutdown resulting from detected problems or LO_2 starvation at the end of the burn duration.

4.1.2.3 Turbomachinery

4.1.2.3.1 Oxidizer Turbopump Hardware Description

The mechanical description of the features of this turbopump is the same as the STBE Derivative Gas Generator oxidizer turbopump. The oxidizer turbopump is shown in Figure 4.1.2.3-1.

4.1.2.3.2 Fuel Turbopump Hardware Description

The fuel turbopump shown in Figure 4.1.2.3-2, is configured as a single-stage centrifugal shrouded impeller pump with an inlet inducer and is driven by a two-stage axial flow turbine. The inducer and impeller, made of fine grained and hot isostatically pressed (HIP) cast Inconel 718, is coupled to the turbine through a single turbine disk with an integral shaft made of Waspaloy. Pump and turbine inlet and discharge housings are fabricated from fine grained cast and HIP Inconel 718 to minimize machining costs. Turbine blades and vanes are made from cast Mar-M-247 nickel alloy. The ball and roller bearings, made of 440C material, will be used to support the pump rotor system. Investigations are ongoing to find an alternate cyrogenic bearing material or combination. Any data and technology obtained through this investigation on the SSME-ATD program will be applied to the fuel pump bearings.

The rotor thrust balance system is accomplished by a dynamic pressure (load) axial balance by properly locating the labyrinth seals at specific diameters. Transient thrust loads are resisted by a forward stop on the rotor contacting a similar stationary stop on the housing, and by a rearward stop located on the ball bearing outer race carrier.

The ball bearing is cooled by first-stage discharge pressure that is controlled leakage flow from the backside of the impeller, through the bearing, then recirculated to the inducer inlet through a controlling labyrinth seal and a hole in the shaft. The roller bearing coolant supply is tapped off the back side of the impeller through an internal passage which provides coolant to the bearing. Between the roller bearing and the turbine, a diaphragm type lift-off seal (similar to the ATD fuel turbopump) is incorporated to prevent cooldown flow from entering the turbine during the pre-start sequence. At engine start, pump pressure increases so as to deflect the lift-off seal to permit flow through the bearing and into the turbine for additional cooling requirements.

The remaining mechanical descriptions of the features of this turbopump are the same as the STBE Derivative Gas Generator fuel turbopump.

4.1.2.3.3 Boost Pump Trade Study

The effects of different boost pump configurations on the main stage pump design and on the performance of a LO_2/CH_4 gas generator engine were evaluated. The pumps studied were a conventional boost, a jet boost and a low/high speed boost pump.

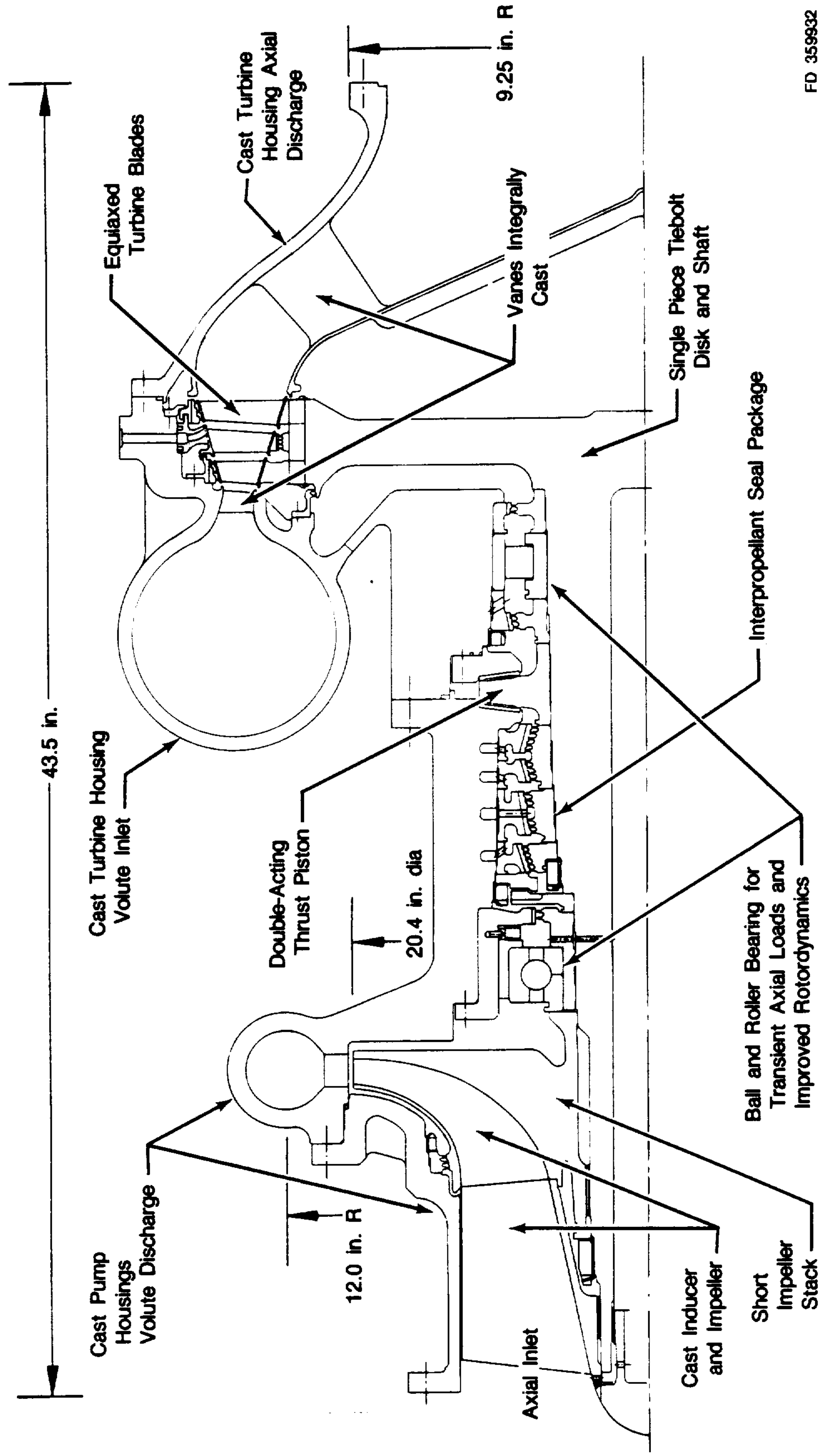


Figure 4.1.2.3-1. STBE Unique Gas Generator LO_2 Turbopump

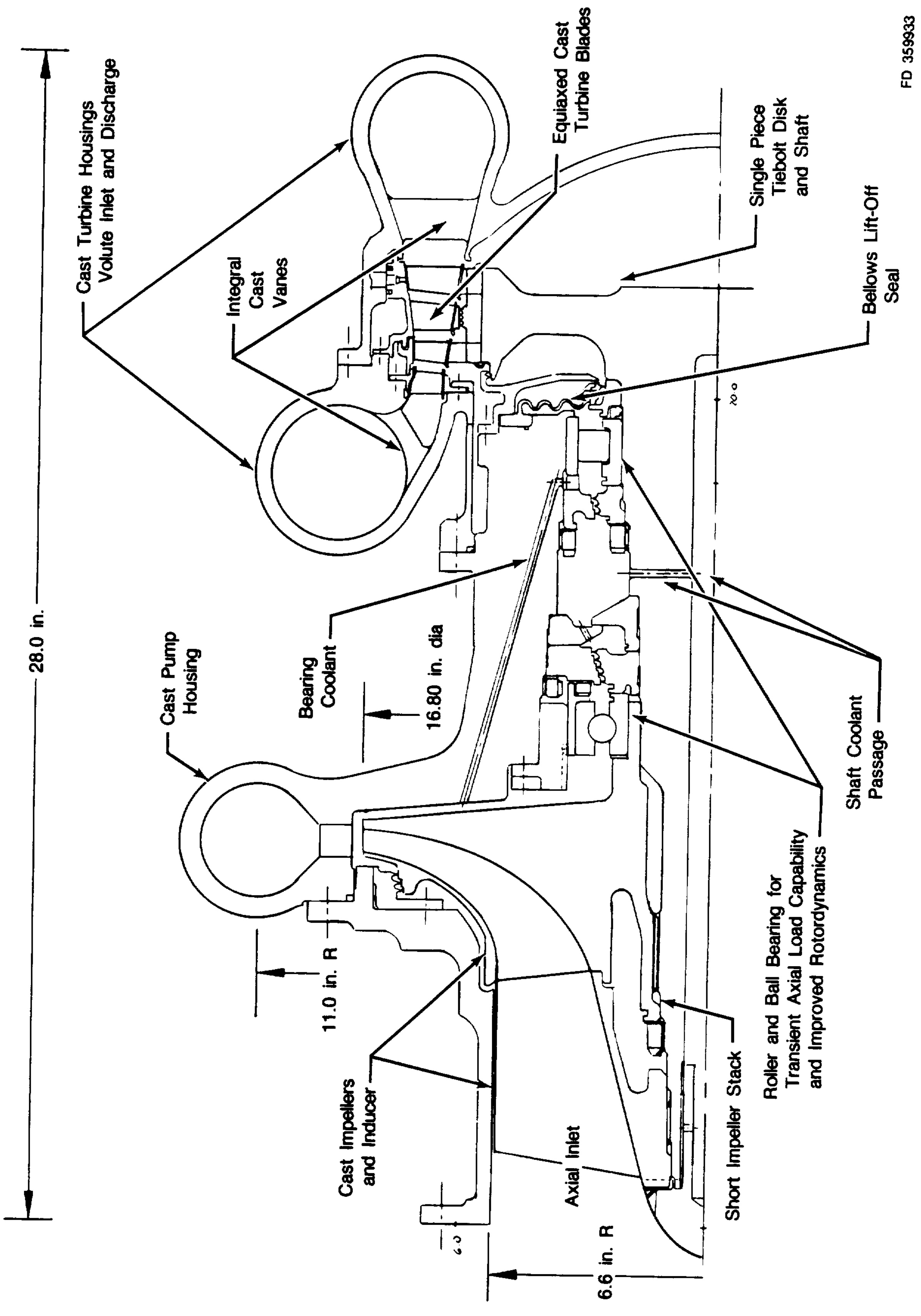


Figure 4.1.2.3-2. STBE Unique Gas Generator Fuel Turbopump

The effects of the boost pumps were seen in the size, weight, turbopump diameter and performance of the main pump. Boost pumps decrease the diameter to the main pump and the number of stages, and also increases the speed of the main pump. Figures 4.1.2.3-3 through -10 summarize the trends on the main pump.

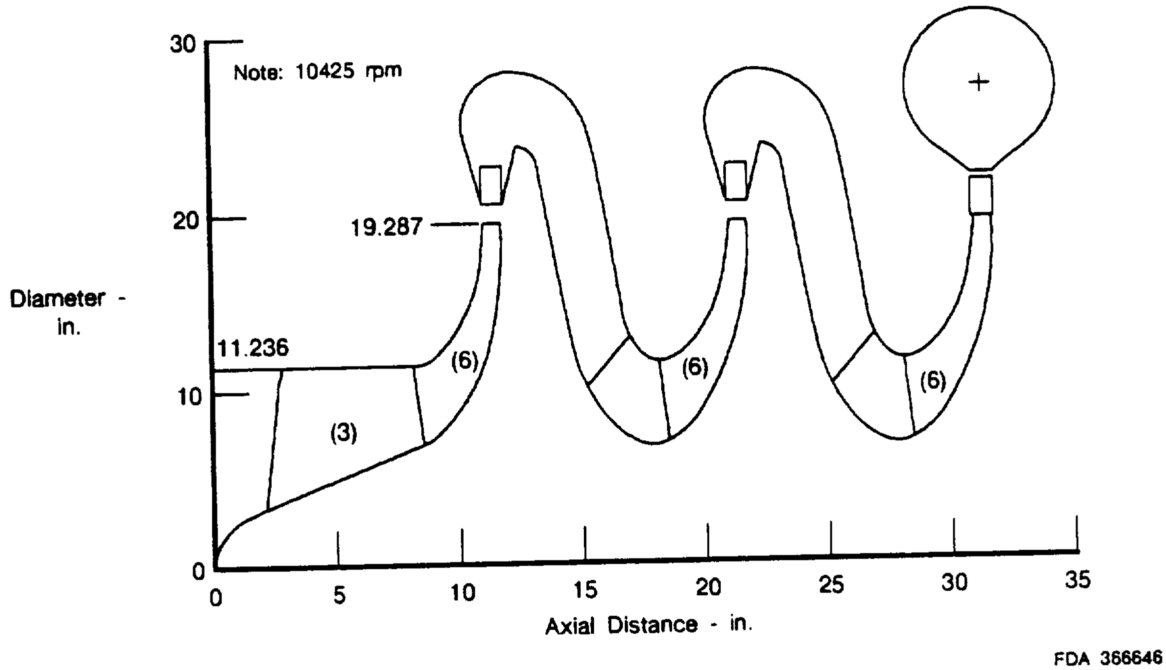


Figure 4.1.2.3-3. STBE Unique Gas Generator Three-Stage Fuel Pump and a Single Discharge Volute

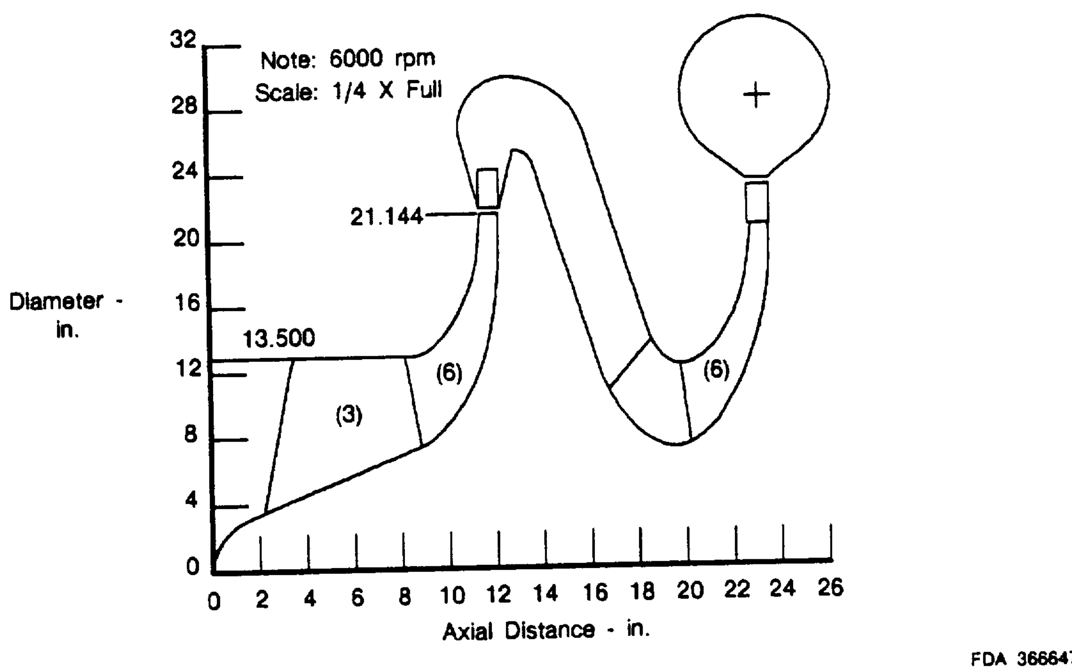


Figure 4.1.2.3-4. STBE Unique Gas Generator Two-Stage Fuel Pump and a Single Discharge Volute

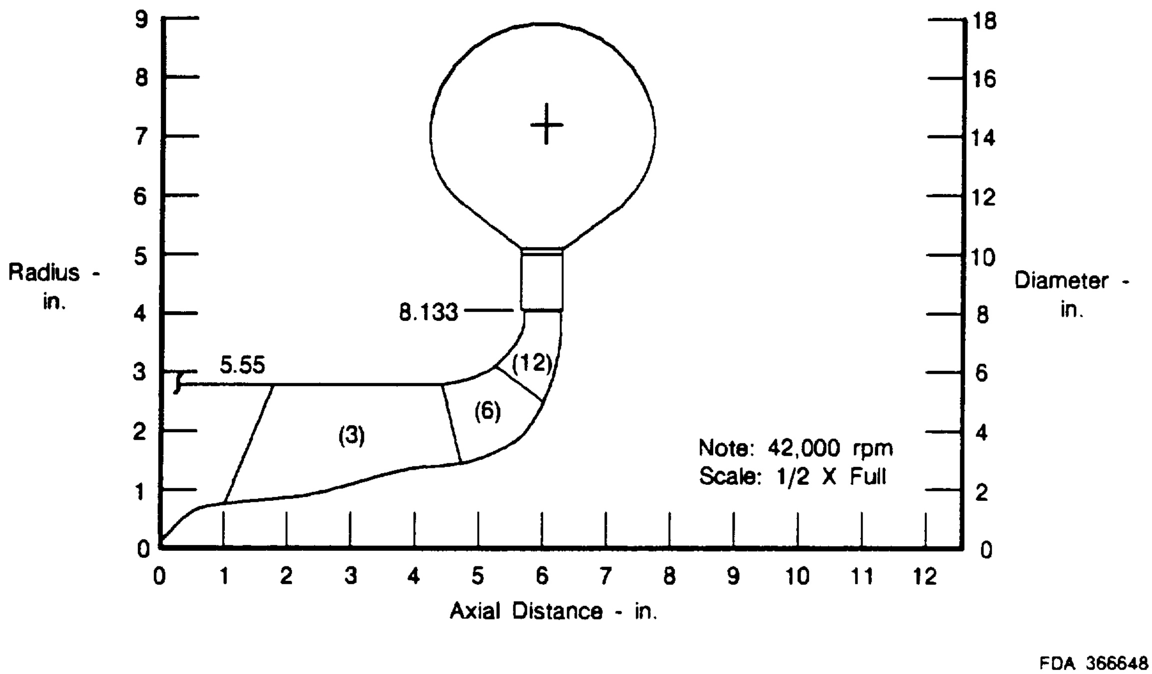


Figure 4.1.2.3-5. STBE Unique Gas Generator Single-Stage Fuel Pump With Boost Pump (Conventional) and a Single Discharge Volute

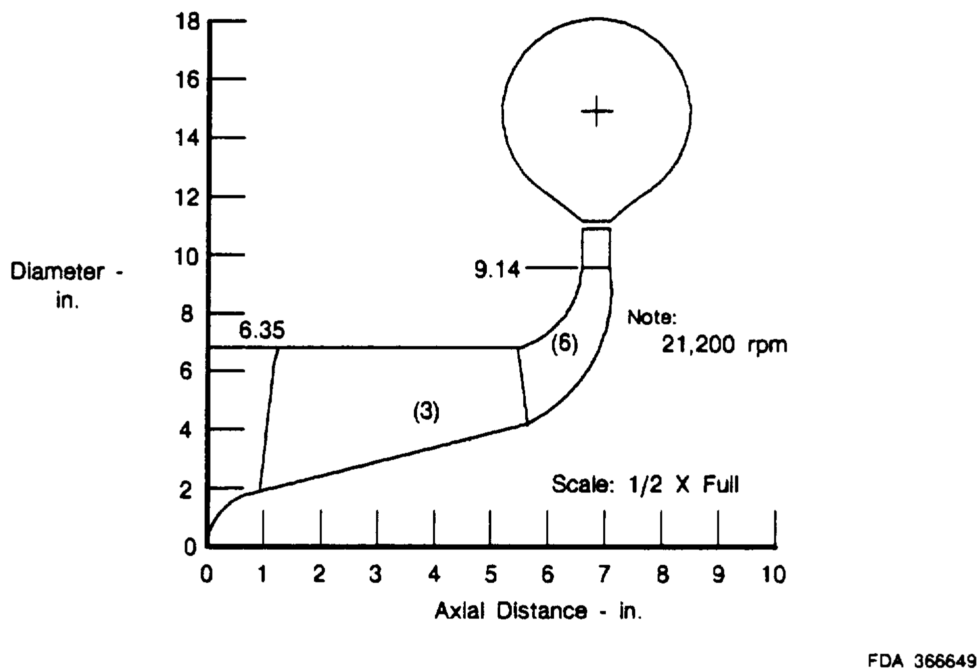


Figure 4.1.2.3-6. STBE Unique Gas Generator Single-Stage Fuel Pump With Boost Pump (Conventional) and a Double Discharge Volute

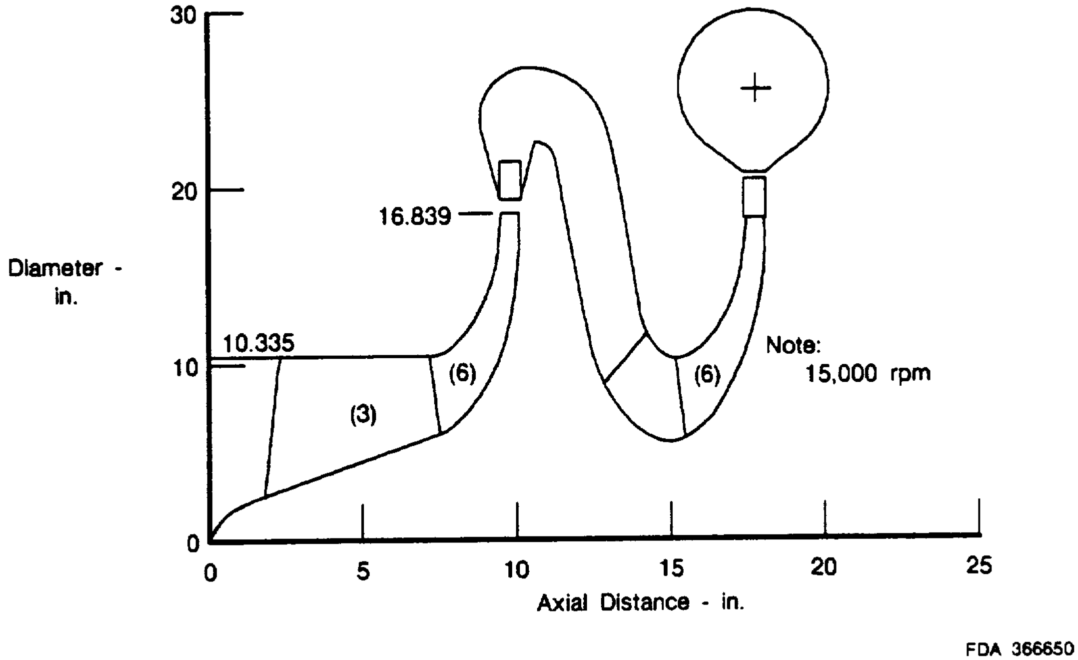


Figure 4.1.2.3-7. *STBE Unique Gas Generator Two-Stage Fuel Pump With Jet Boost Pump and a Single Discharge Volute*

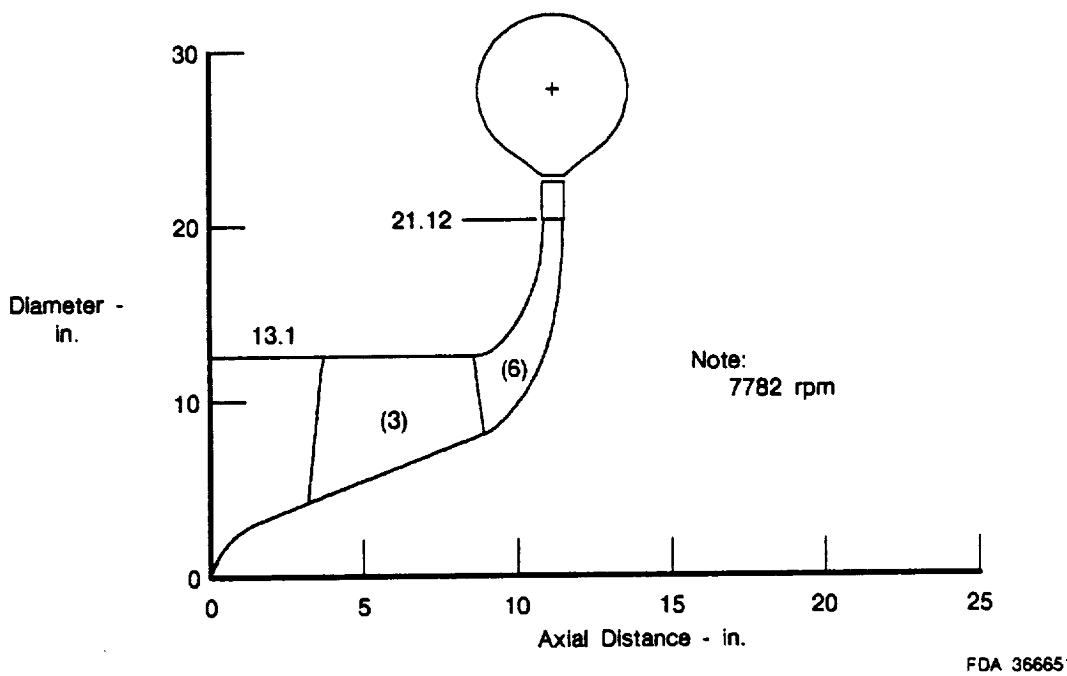


Figure 4.1.2.3-8. *STBE Unique Gas Generator Single-Stage Fuel Pump With Jet Boost Pump and a Single Discharge Volute*

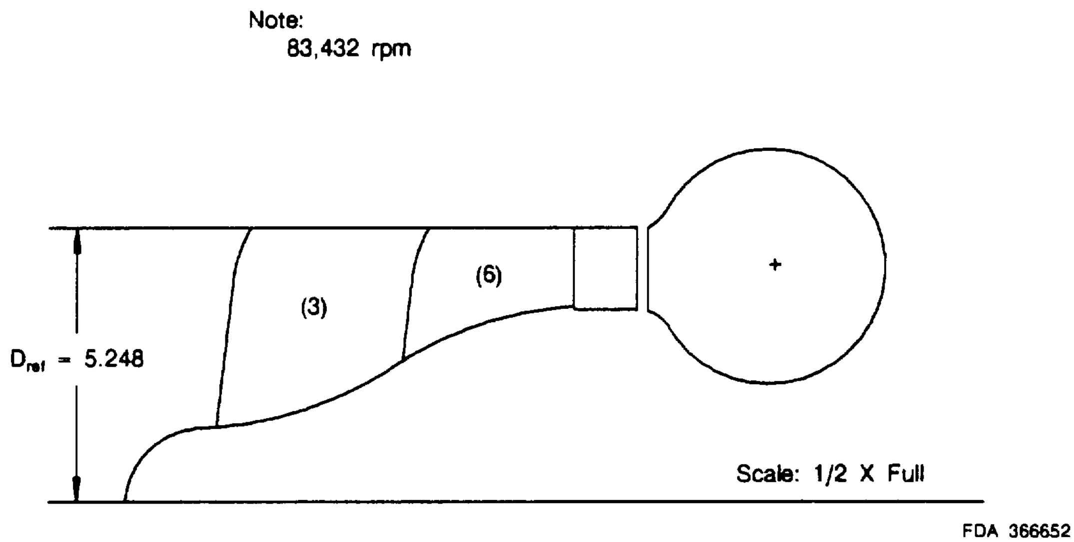


Figure 4.1.2.3-9. STBE Unique Gas Generator High-Speed Boost Pump and a Single Discharge Volute

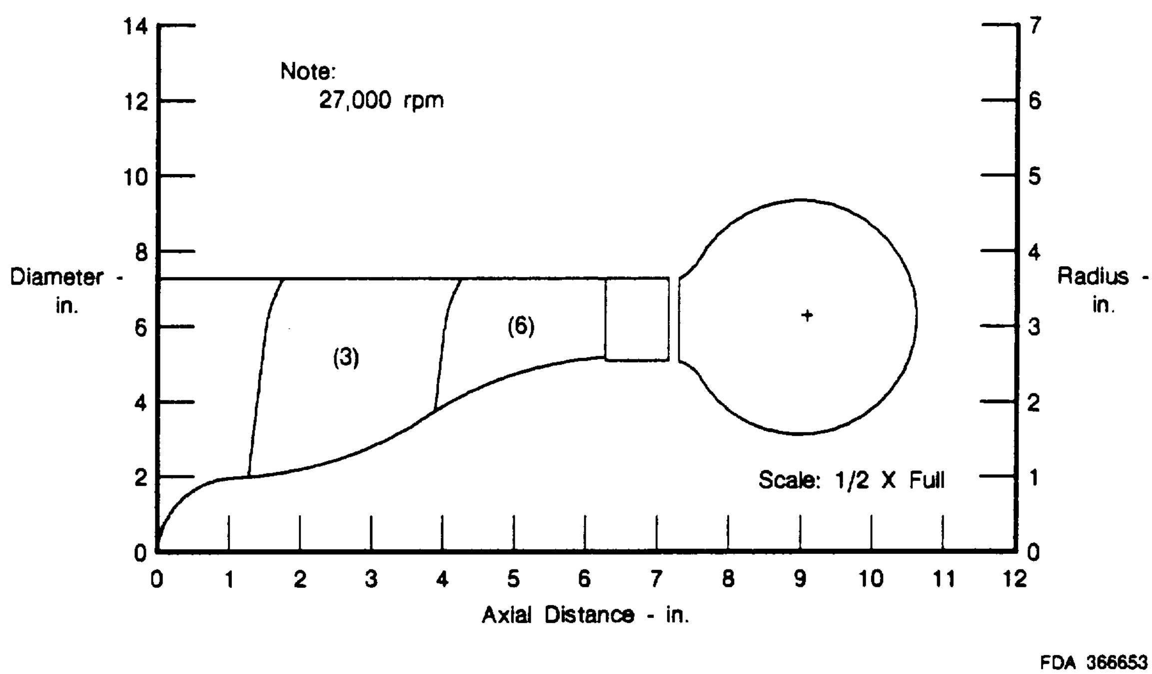


Figure 4.1.2.3-10. STBE Unique Gas Generator High-Speed Boost Pump and a Single Discharge Volute

4.1.2.4 Combustor

The unique STBE minimum chamber volume, injector design, and acoustic liner design were determined using the procedures outlined in section 4.1.1.4 for the derivative STBE engine. The unique STBE chamber and injector element design are summarized in Table 4.1.2.4-1. The characteristic length (L*) given in the table (31.3 inches) is the minimum required to meet the 98.0 percent characteristic velocity efficiency specified for the engine. Note that the fuel and LO_2

pressure drops are less than in the derived STBE core since the flow areas could be set for STBE operating conditions alone.

Table 4.1.2.4-1. *Unique STBE Combustor and Injector Design*

Chamber L* (Min)-in.	31.3
Fuel Flow-lb/sec	502.1
ΔP Fuel-psi	168.0
LO_2 Flow-lb/sec	1816.3
ΔP LO_2 psi	167.3
No. of Elements	395
Element ID-in.	0.366
Annular Gap-in.	0.021

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The acoustic liner design set for the unique STBE is given in Table 4.1.2.4-2. This liner will provide a 30 percent acoustic absorption at the first tangetial frequency (1212 Hz) of the combustion chamber.

Table 4.1.2.4-2. *Acoustic Liner Design*

Chamber Pressure-psi	2396
Aperture — Gas Temperature-°R	2000
Aperture — Gas Molecular Wt.	22.4
Hole Diameter-in.	0.10
Hole Length-in.	0.35
Area Ratio	0.05
Backing Cavity Depth-in.	0.6
Liner Length-in.	4.0

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4.1.2.4.1 Main Injector

The mechanical description of the features of this main injector are the same as the STBE Derivative Gas Generator main injector. Figure 4.1.2.4-1 shows the main injector, and Figures 4.1.2.4-2 and -3 show the injector element configuration and injector pattern, respectively.

4.1.2.4.2 Combustion Chamber

The combustion chamber is regeneratively cooled by fuel from the high pressure pump discharge. The fuel enters the thermal skin cooling jacket at the interface of the regeneratively cooled nozzle manifold. The coolant then flows forward, counter to the gas path flow, to the throat. The fuel cools the chamber wall, exits at the injector interface internal manifold, and enters the injector. This flow configuration provides the coolest fuel at the throat where wall heat flux is highest. The combustion chamber is shown in Figure 4.1.2.4-4.

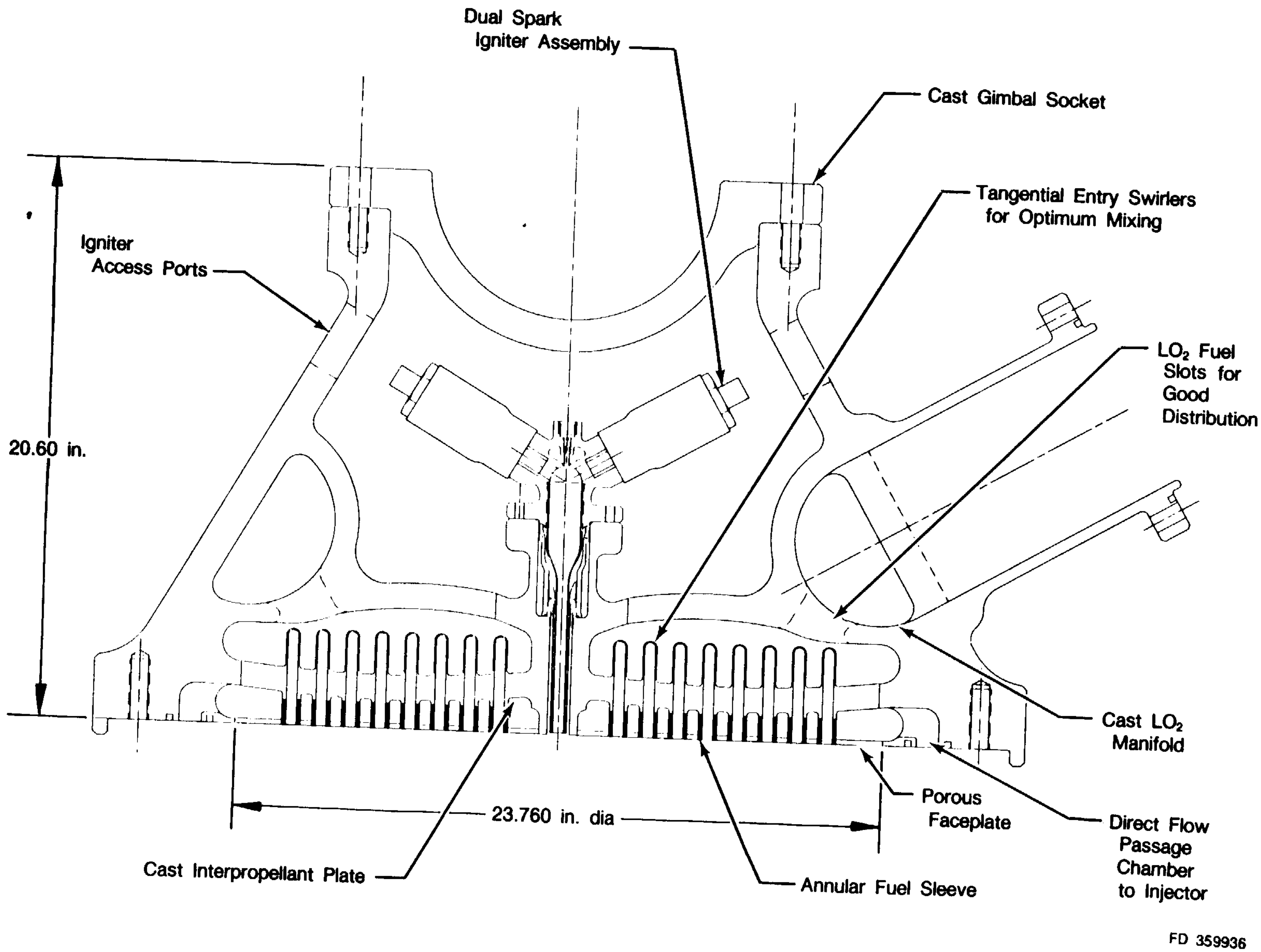


Figure 4.1.2.4-1. STBE Unique Gas Generator Main Injector

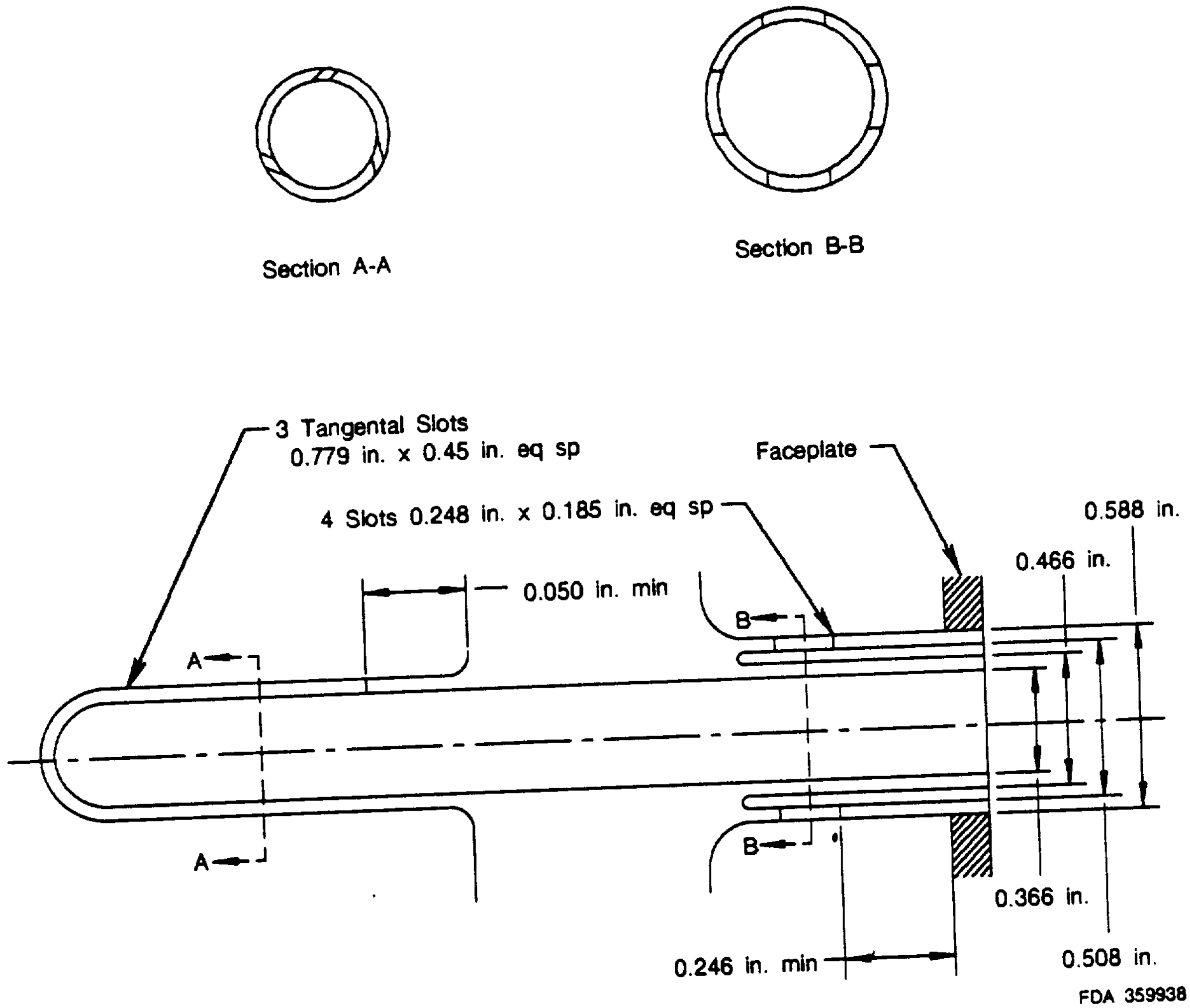


Figure 4.1.2.4-2. STBE Unique Gas Generator Main Injector Element

The STBE unique gas generator thrust chamber features a machined passage thermal-skin NASA-Z liner/nickel closeout assembly surrounded by a structural jacket. The chamber inlet manifold is common with the tubular nozzle which improves the inlet geometry and reduces inlet pressure drop. The coolant enters the common inlet manifold and counterflows toward the injector where it discharges directly into the injector. Since the chamber is cooled with the entire chamber flow, the exit manifold can be eliminated to reduce the coolant exit pressure drop. The STBE unique gas generator has a throat diameter of 15.04 inches, an injector diameter of 23.79 inches and a contraction ratio of 2.5.

The coolant passage dimensions are sized to meet the heat transfer and cycle requirements at the 750K lbf vacuum thrust at 2250 psia chamber pressure design point and reflect the following design guidelines.

- Liner wall thickness > 0.35 in.
- Passage aspect ratio < 5.0.
- Passage land width > 0.050 in.
- Cooling enhancement from passage curvature.
- Coolant Mach number < 0.5.

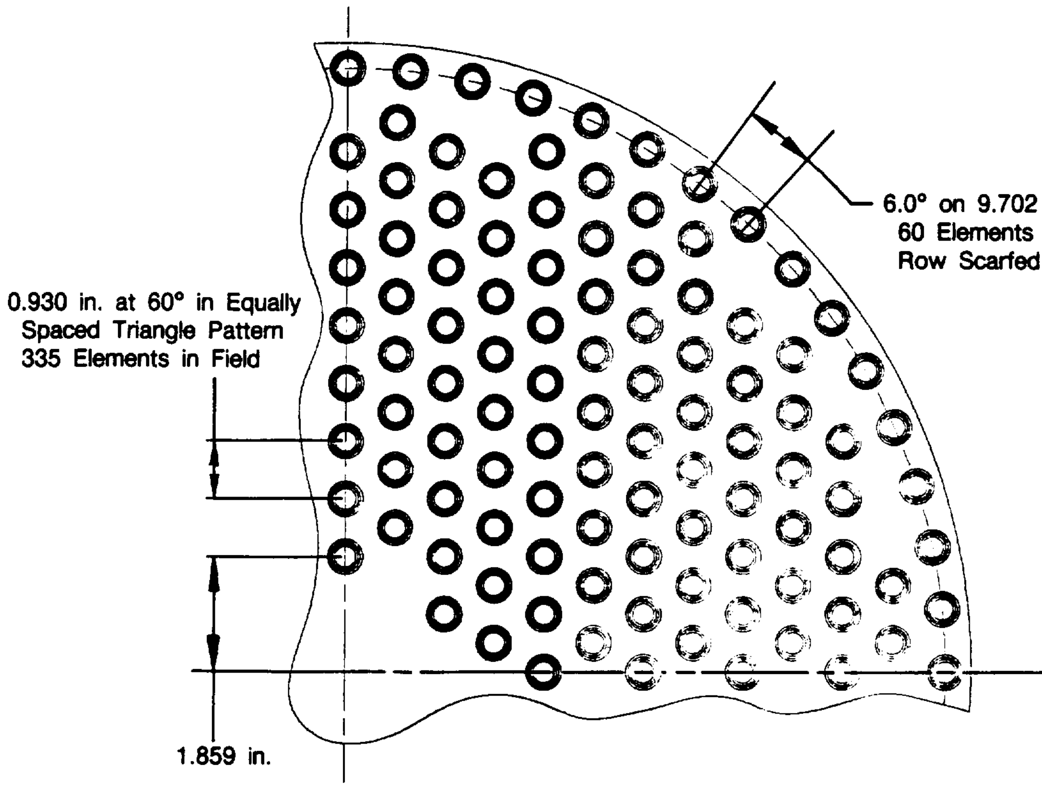


Figure 4.1.2.4-3. STBE Unique Gas Generator Main Injector Pattern

Figure 4.1.2.4-5 summarizes the throat chamber contour and tube geometry.

The hydrogen coolant enters the liner at 236 R and 4934 psia and exits at 430 R and 2589 psia. The maximum predicted values of hot wall temperature and heat flux are 1530 R and 61.5 Btu/in.^{2}-sec, respectively. The flux highest calculated coolant Mach number is 0.2. The enhancement of the coolant side heat transfer coefficient at the maximum heat flux location is approximately 35 percent. Figure 4.1.2.4-6 summarizes the predicted thermal performance characteristics for the thrust chamber.

The mechanical description of the features of this combustion chamber are the same as the STBE Derivative Gas Generator combustion chamber.

4.1.2.4.3 Torch Igniter — Gas Generator

A continuous burning torch igniter was chosen for use in both the gas generator and main combustion systems because of the simplicity of the design and reliability in tests. The igniter configuration employed evolved from development efforts since 1957 at Pratt & Whitney and is based on experience gained from the successful RL10 and XLR-129 engine programs.

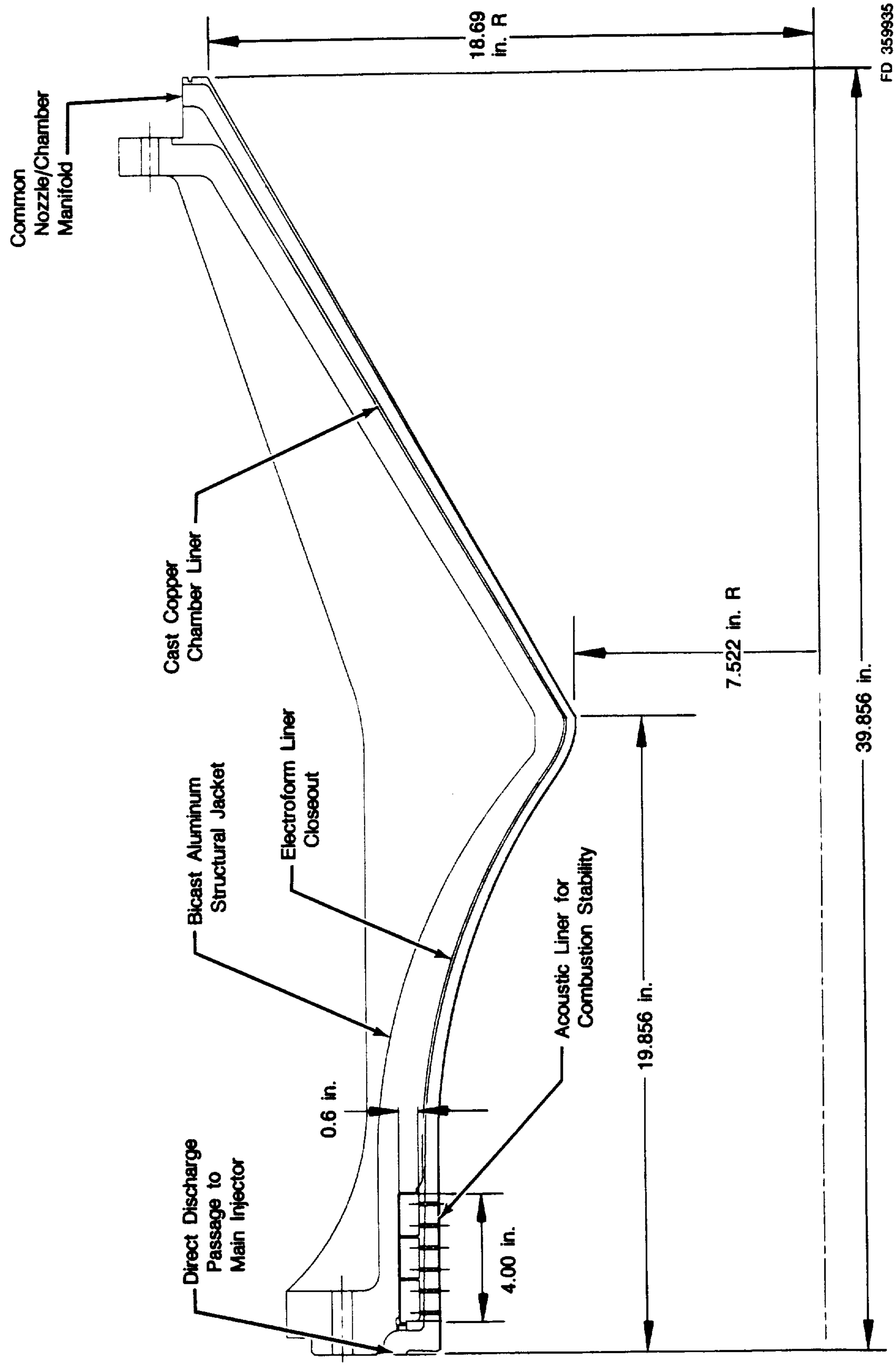
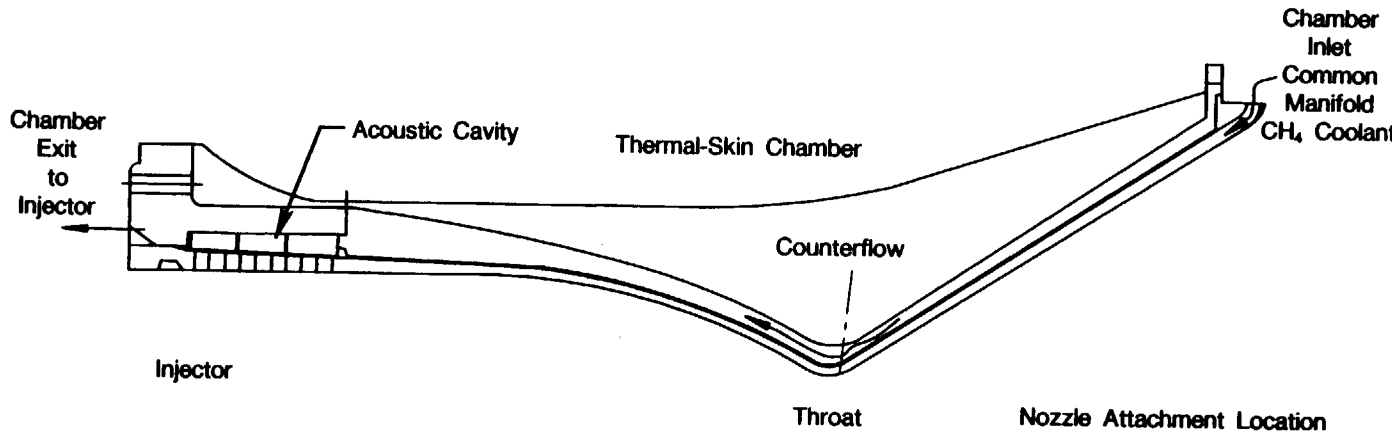


Figure 4.1.2.4-4. STBE Unique Gas Generator Combustion Chamber



Chamber Contour Data

Chamber Length = 20 in.
Divergent Nozzle Length = 20 in.
Throat Diameter = 15.04 in.
Injector Diameter = 23.79 in.
Contraction Ratio = 2.5
Divergent Nozzle Area Ratio = 5.48
L^* = 42.0 in.
η_{c^*} (Throat) = 0.98
Number of Passages = 370
Liner Construction - Thermal-Skin
Liner Material - NASA Z

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	Passage Width (in.)	Passage Height (in.)	Land Width (in.)	Wall Thickness (in.)
-20.0	11.89	0.081	0.394	0.121	0.035
-16.1	11.89	0.081	0.374	0.121	0.035
-13.8	11.89	0.090	0.314	0.111	0.035
-12.7	11.88	0.090	0.305	0.111	0.035
-9.2	11.43	0.090	0.283	0.102	0.035
-5.0	10.34	0.090	0.238	0.085	0.035
-3.7	9.19	0.068	0.220	0.089	0.035
-0.0	7.52	0.068	0.210	0.060	0.035
2.7	9.05	0.068	0.280	0.086	0.044
5.0	10.33	0.120	0.349	0.066	0.051
10.8	13.55	0.120	0.488	0.109	0.070
20.0	18.69	0.120	0.488	0.193	0.100

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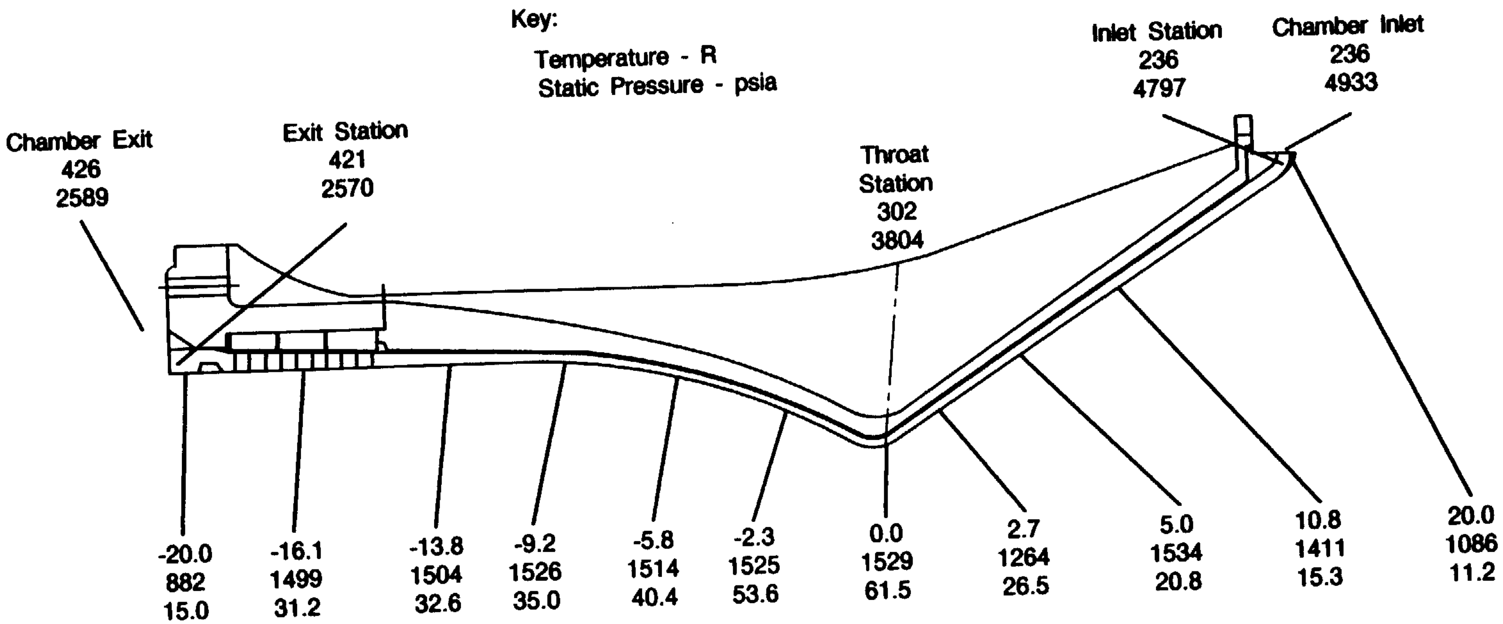
Figure 4.1.2.4-5. STBE Unique Gas Generator Chamber Cooling Design Configuration



Coolant Performance

Thrust = 120%
M_{cool} = 502.1 lbm/ sec

Chamber Heat Transfer Performance

Thrust - lbf	750K	625K
Chamber Pressure - psia	2400	2054
Coolant Flow - lbm/sec	502.1	439.9
Inlet Temperature - R	236.0	229.0
Exit Temperature - R	430.0	418.0
Coolant Heat Pickup - Btu/sec	83362.0	74143.0
Inlet Pressure - psia	4933.0	3985.0
Exit Presure - psia	2589.0	2146.0
Pressure Drop - psid	2344.0	1383.0

Hot Wall Temperature & Heat Flux

Key:

Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2} - sec

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Figure 4.1.2.4-6. STBE Unique Gas Generator Chamber Heat Transfer Performance Summary

In the gas generator, the torch is mounted in the combustor wall, two inches axially from the injector face, and expels the hot torch combustion gases at a right angle to the flow path from the gas generator injector, thus providing safe, efficient, reliable ignition of the combustion system. In the main combustion chamber, the torch is mounted axially in the center of the injector, directing the torch down along the centerline of the combustion chamber.

The construction of the torch assembly is discussed in Space Transportation Main Engine Configuration Study P&W FR-19830-1, Volume II, page 93.

4.1.2.4.4 Unique STBE Gas Generator Combustion System

The mechanical description of the features of this gas generator combustion system are the same as the STBE Derivative Gas Generator combustion system. Figures 4.1.2.4-7 through -10 present the gas generator assembly, injector element and injector pattern design.

4.1.2.4.5 Mixture Ratio Trade Study

An engine heat transfer trade study was performed to determine the effects of engine mixture ratio on thrust chamber and tubular nozzle performance in a LO_2/CH_4 gas generator engine. Inlet mixture ratios of 3.0 and 3.57 were evaluated. The effects of increased mixture ratio on thrust chamber performance are a decrease in chamber flow rate and an increase in the pressure entering the inlet manifold. This increase in mixture ratio resulted in 14.5 percent increase in chamber heat transfer rate.

The effects of increased mixture ratio on nozzle performance are a decrease in manifold inlet pressure, a decrease in nozzle coolant flow rate and an increase in temperature at the exit manifold. The increase in mixture ratio resulted in an 11.5 percent pickup in heat transfer rate. Figure 4.1.2.4-11 summarizes these trends.

4.1.2.5 Nozzle

4.1.2.5.1 Unique STBE Gas Generator Regeneratively Cooled Nozzle

The regeneratively cooled nozzle is constructed from 990 SPIF (Super Plastic Inflation Formed) tubes of AISI 347 stainless steel, surrounded by a structural shell of closed cell elastomeric foam with a filament wound composite overwrap. This shell is also designed to carry all hoop loads. The regeneratively cooled nozzle is shown in Figure 4.1.2.5-1.

The mechanical description of the features of this nozzle are the same as the STBE Derivative Gas Generator Regeneratively Cooled Nozzle.

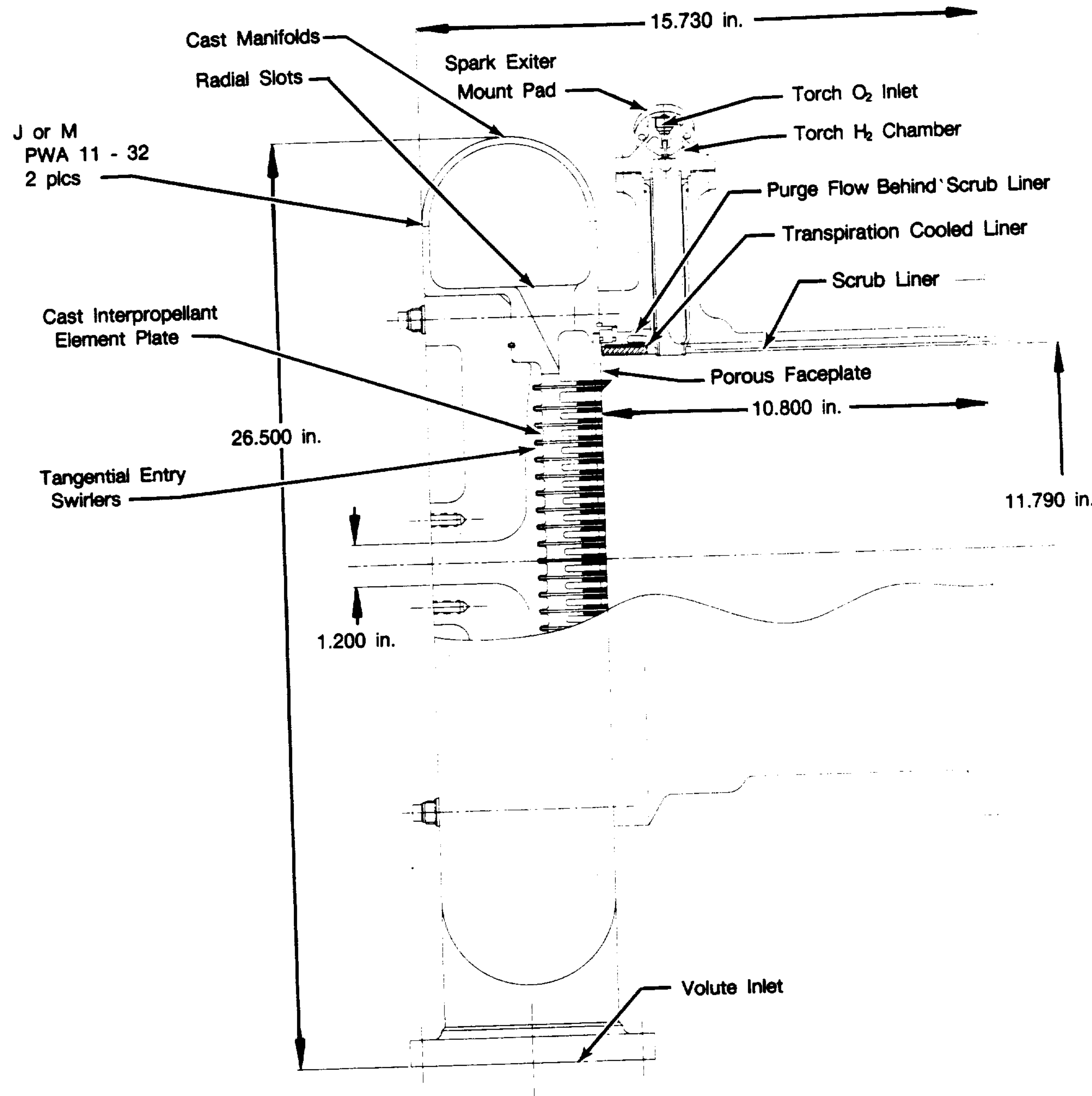


Figure 4.1.2.4-7. STBE Unique Gas Generator Assembly

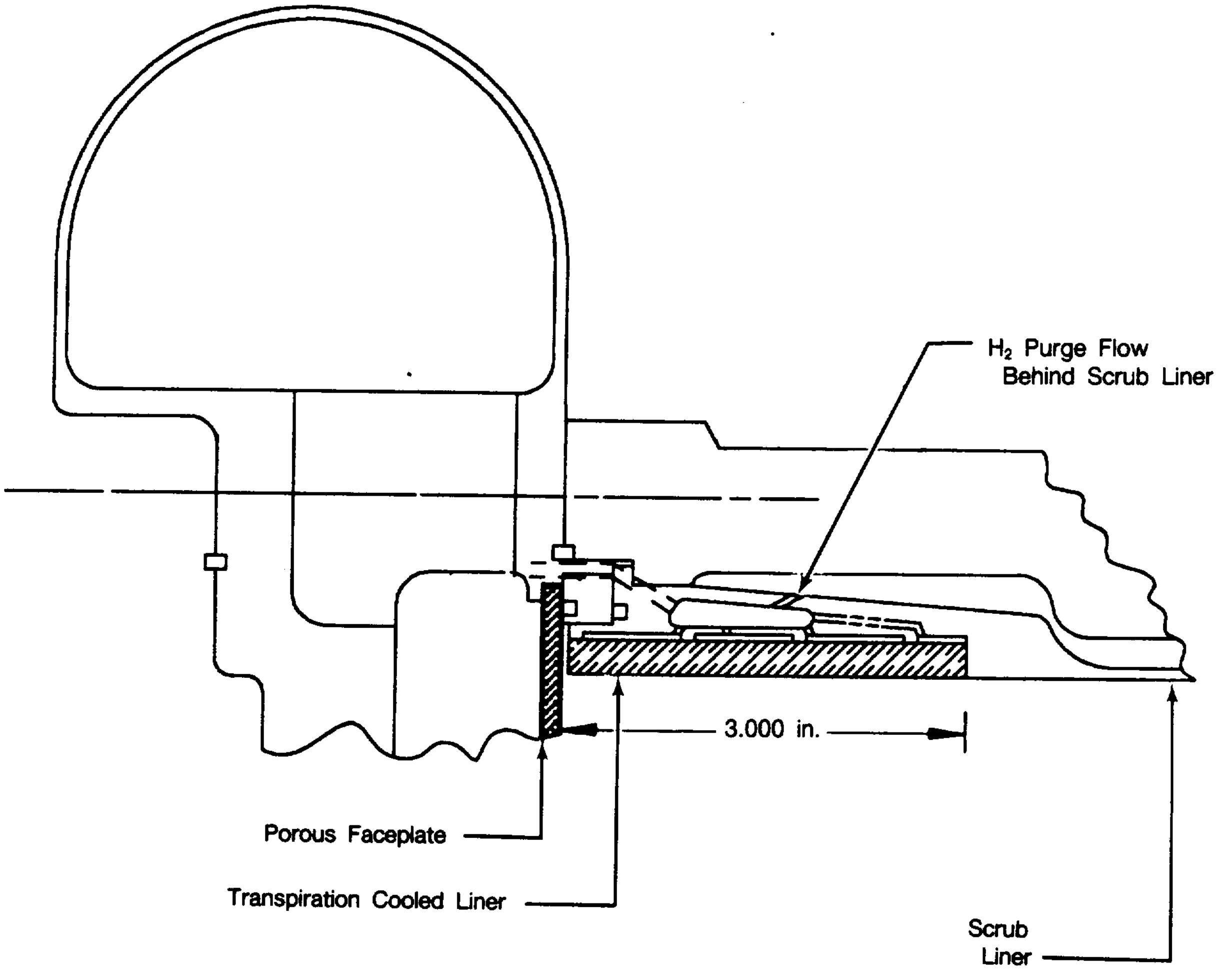


Figure 4.1.2.4-8. STBE Unique Gas Generator Assembly

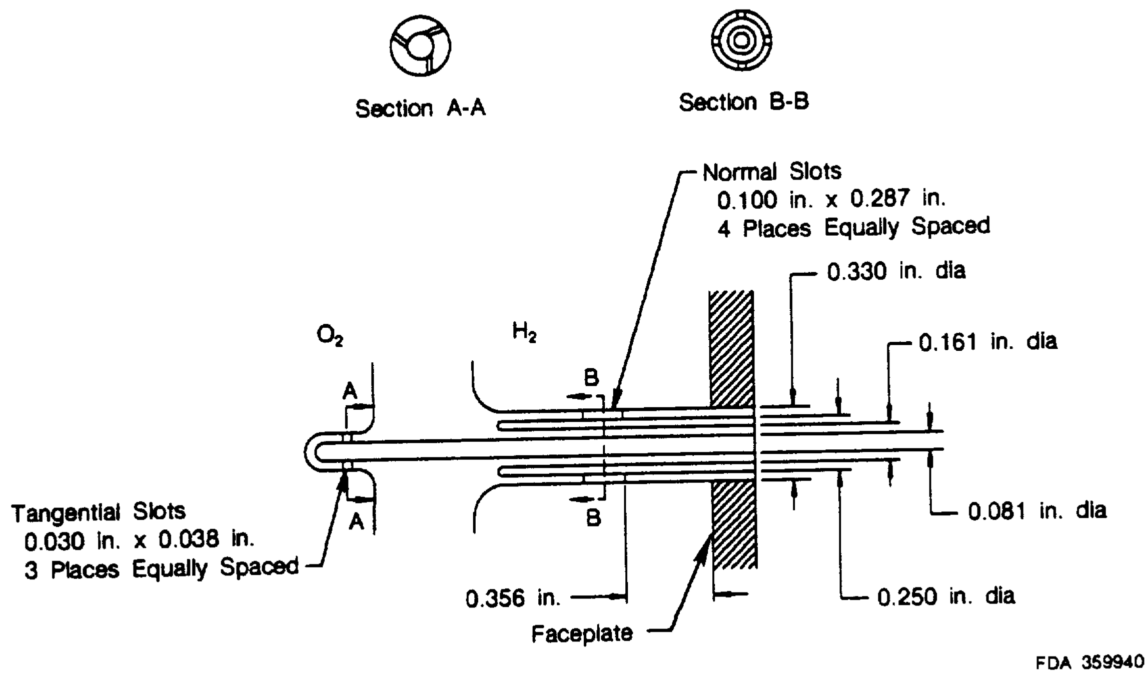


Figure 4.1.2.4-9. STBE Unique Gas Generator Assembly Injector Element

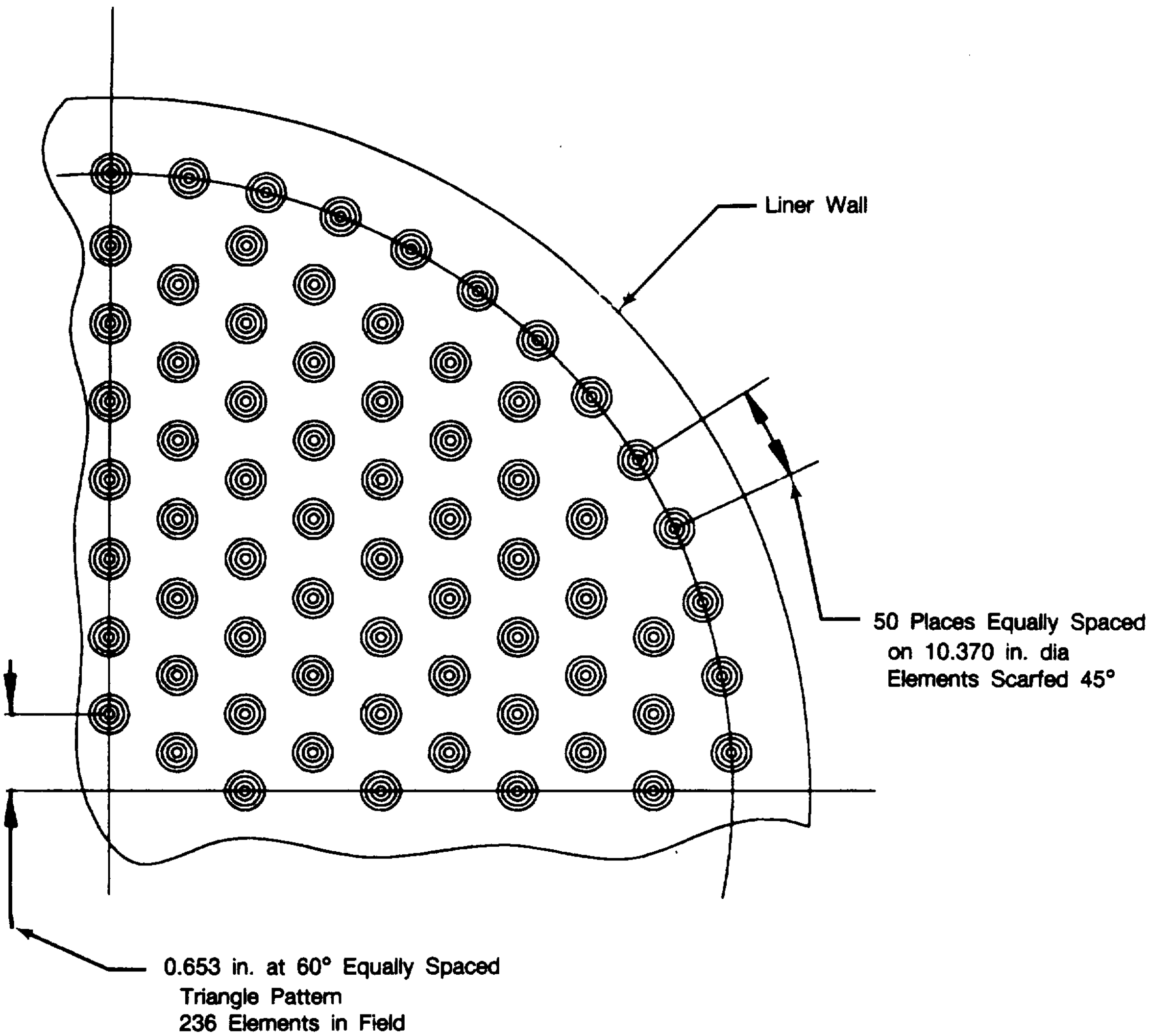


Figure 4.1.2.4-10. STBE Unique Gas Generator Assembly Injector Pattern

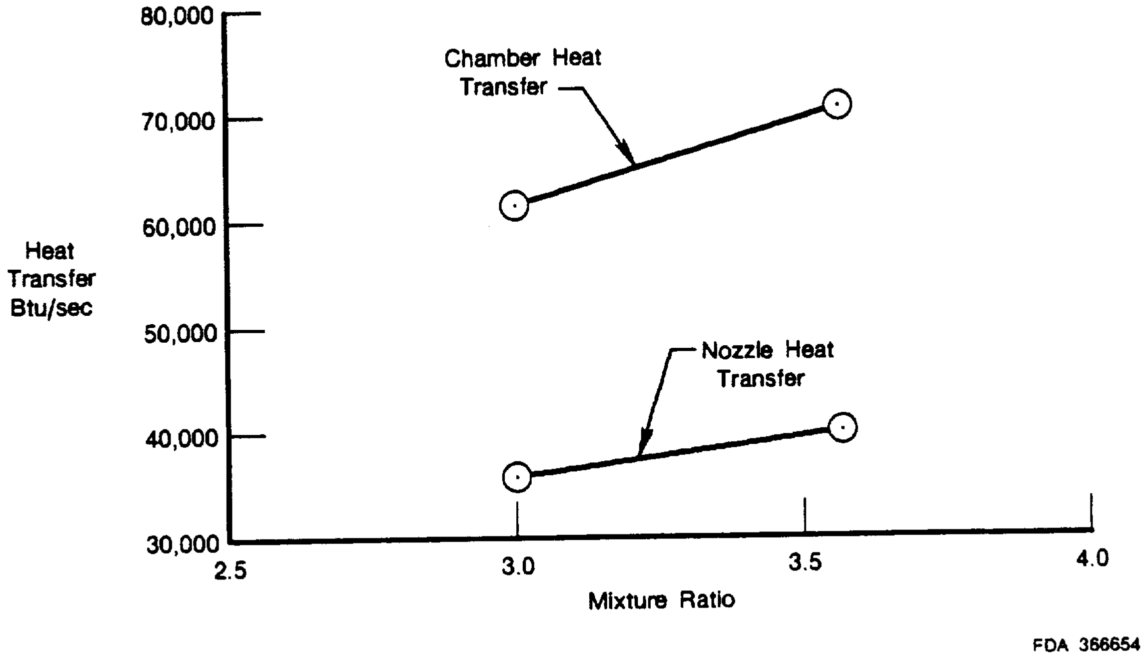


Figure 4.1.2.4-11. Effect of Mixture Ratio on Chamber and Nozzle Performance

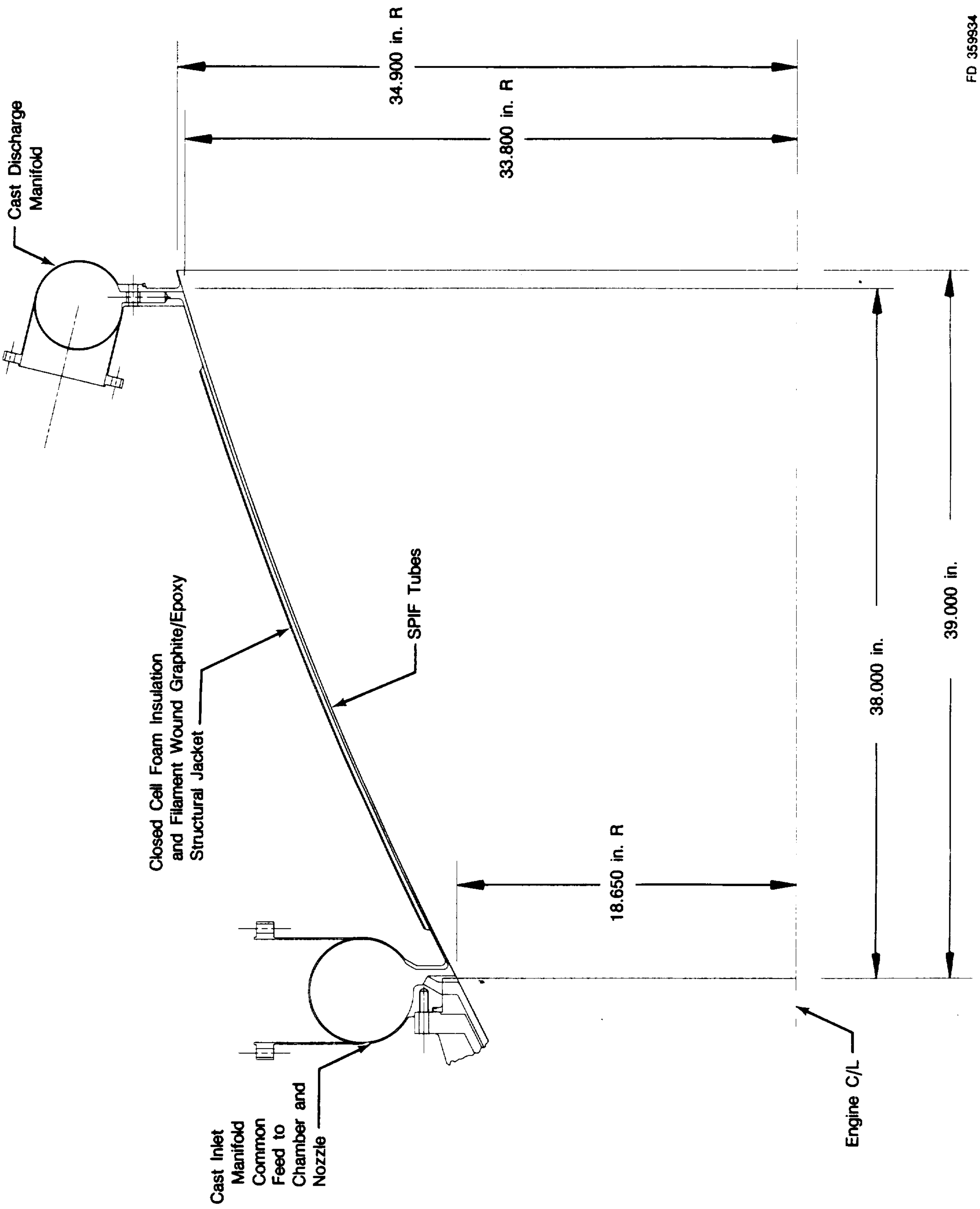


Figure 4.1.2.5-1. STBE Unique Gas Generator Regeneratively Cooled Nozzle

Figure 4.1.2.5-2 summarizes the regeneratively cooled nozzle geometry. The nozzle is constructed of 990 super plastic inflation formed AISI 347 stainless steel passages that simulate tubes. The nozzle has a length of 38 inches, an inlet expansion area ratio of 5.48:1 and an exit expansion area ratio of 20:1. The nozzle passage dimensions are sized to meet the heat transfer and cycle requirements at the 750K lbf vacuum thrust at 2250 psia chamber pressure design point and reflect the following design guidelines.

- Maximum stress < 0.2 percent yield strength.
- Ultimate tube temperature margin > 375 R.
- Coolant Mach number < 0.5.
- Wall thickness > 0.013 in.
- Wall temperature < 2260 R.

The coolant enters the nozzle through the inlet manifold that is common to the thrust chamber and flows parallel to the gas path. The nozzle is cooled with 116 lbm/sec of fuel that enters at 236 R and 4925 psia and exits at 589 R and 4685 psia. The maximum levels of predicted hot wall temperature and heat flux are 1648 R and 8.7 Btu/in.^{2}-sec, respectively. Figure 4.1.2.5-3 summarizes the predicted thermal performance characteristics for the regeneratively cooled nozzle.

4.1.2.5.2 Film and Radiation Cooled Nozzle

The film and radiation cooled nozzle is fed coolant from the LO_2 pump turbine discharge and is supplied to the nozzle through a circumferentially tapered toroidal manifold, which injects the coolant along the inner surface of the nozzle.

The nozzle, shown in Figure 4.1.2.5-4, is constructed of a fine grained cast and HIP Inconel 718 inlet manifold with fabricated toroidal structure (welded sheet metal) and a bolt-on columbium sheet metal nozzle.

The STBE film/radiation cooled nozzle is 60 inches long and extends from an expansion area ratio of 20 to 35. Gas generator discharge flow is introduced as a film at the forward end of the radiation nozzle to provide film cooling. The film provides a thermal barrier between the gas path and nozzle wall, thereby eliminating the need for more complex cooling methods. The highest predicted nozzle wall temperature is 2100 R. Figure 4.1.2.5-5 is a schematic showing the axial distribution of wall temperature and heat flux.

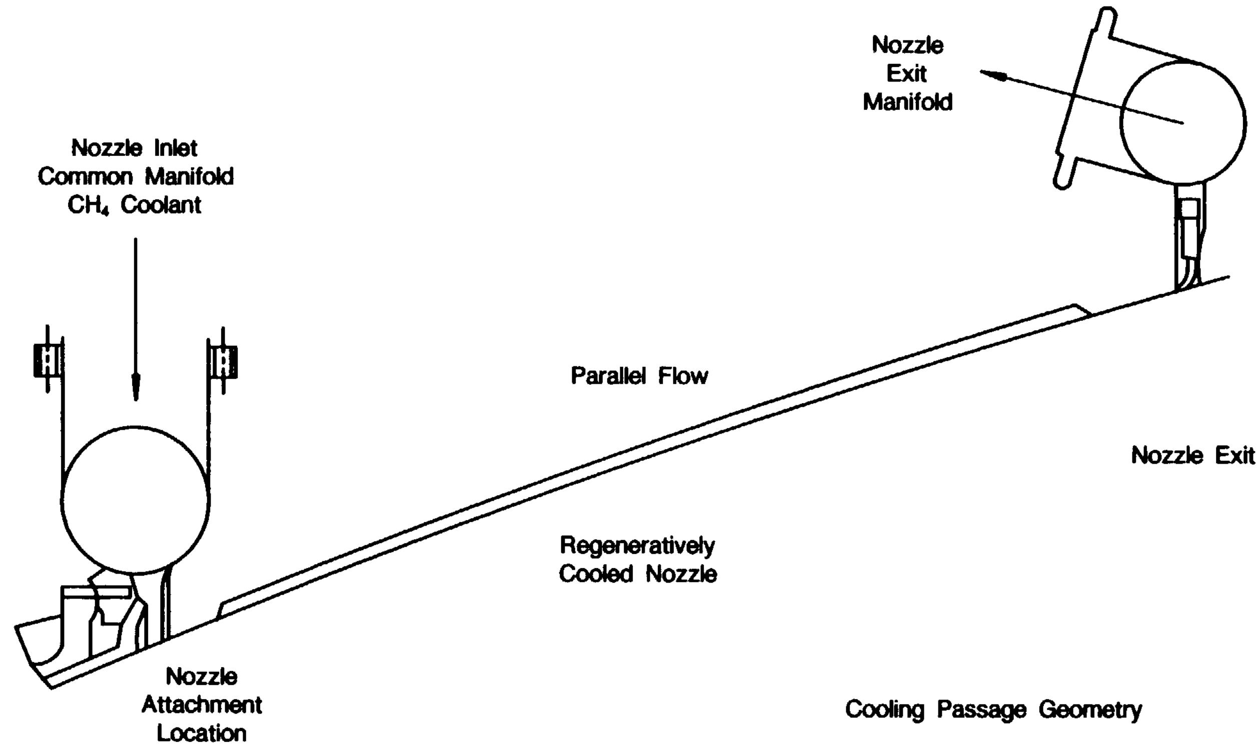
4.1.2.6 Controls

The description of the engine controls for the Unique STBE Gas Generator Engine is the same as the controls for the Derivative STBE Gas Generator Engine.

4.1.2.7 Engine Configuration and Integration

4.1.2.7.1 Unique STBE Gas Generator Engine Assembly

The arrangement of the external configuration of the engine was based on optimization of component accessibility for routine component inspection, removal and replacement operations. Figures 4.1.2.7-1 and -2 show the side and top views of the engine assembly and its major components.



Nozzle Contour Data

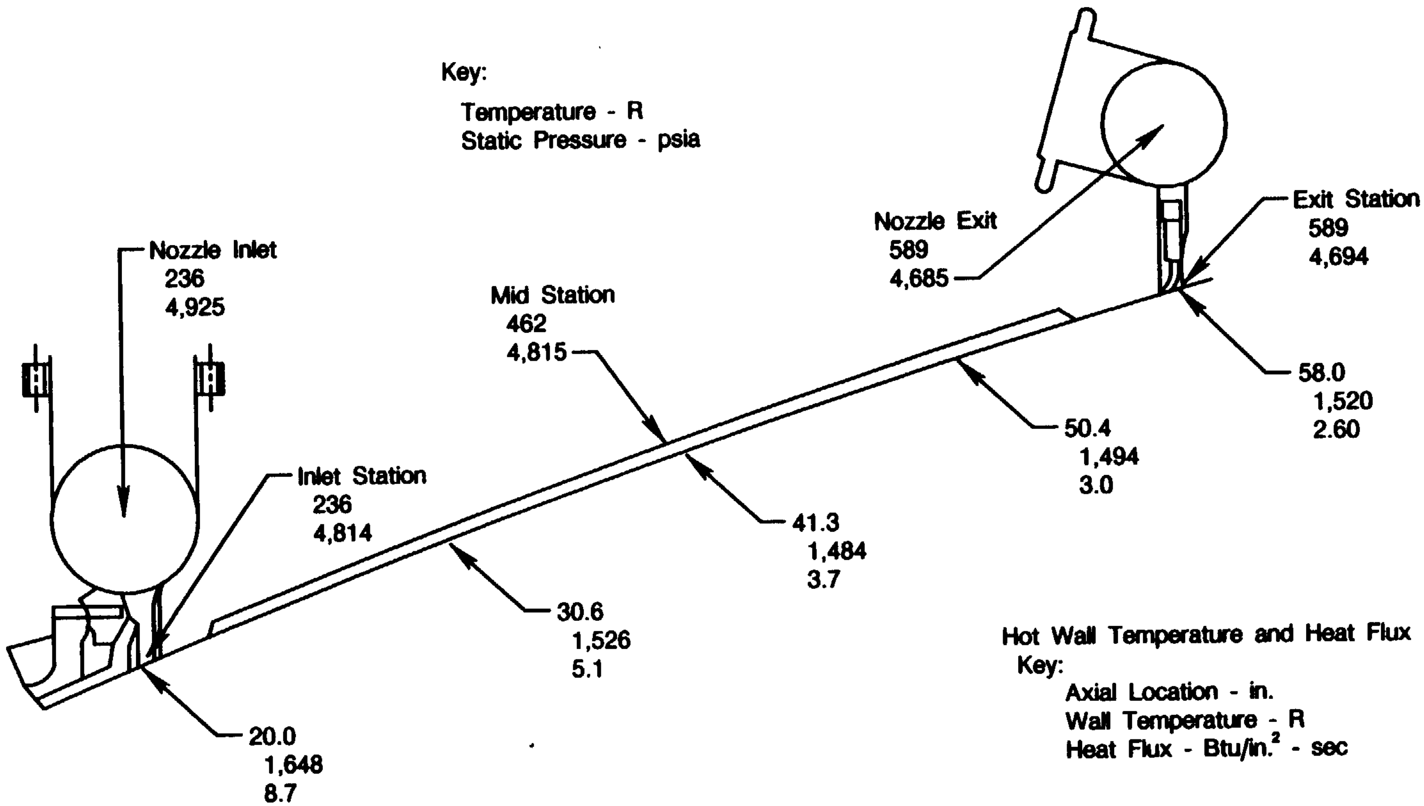
Nozzle Length = 38 in.
Inlet Expansion Ratio = 5.48
Exit Expansion Ratio = 20.0
Number of Passages = 990
Nozzle Construction - Super Plastic Inflation Formed
Nozzle Material - AISI 347

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)
20.0	18.69	0.109	0.112	0.020
30.6	23.71	0.141	0.144	0.020
41.3	28.09	0.169	0.173	0.020
50.4	31.34	0.190	0.194	0.020
58.0	33.80	0.205	0.210	0.020

FDA 363350

Figure 4.1.2.5-2. STBE Unique Gas Generator Nozzle Cooling Design Configuration



Coolant Performance
- Thrust = 120%
- M_{cool} = 116 lbm/sec

Nozzle Heat Transfer Performance

Thrust - lbf	750K	625K
Chamber Pressure - psia	2400	2,054
Coolant Flow - lbm/sec	116.1	97.7
Inlet Temperature - R	236.4	229.3
Exit Temperature - R	589.0	583.7
Coolant Heat Pickup - Btu/sec	33,462.0	29,161.6
Inlet Pressure - psia	4,924.9	3,984.9
Exit Pressure - psia	4,684.9	3,809.4
Pressure Drop - psid	241.0	175.5

FDA 363801

Figure 4.1.2.5-3. STBE Unique Gas Generator Nozzle Heat Transfer Performance Study

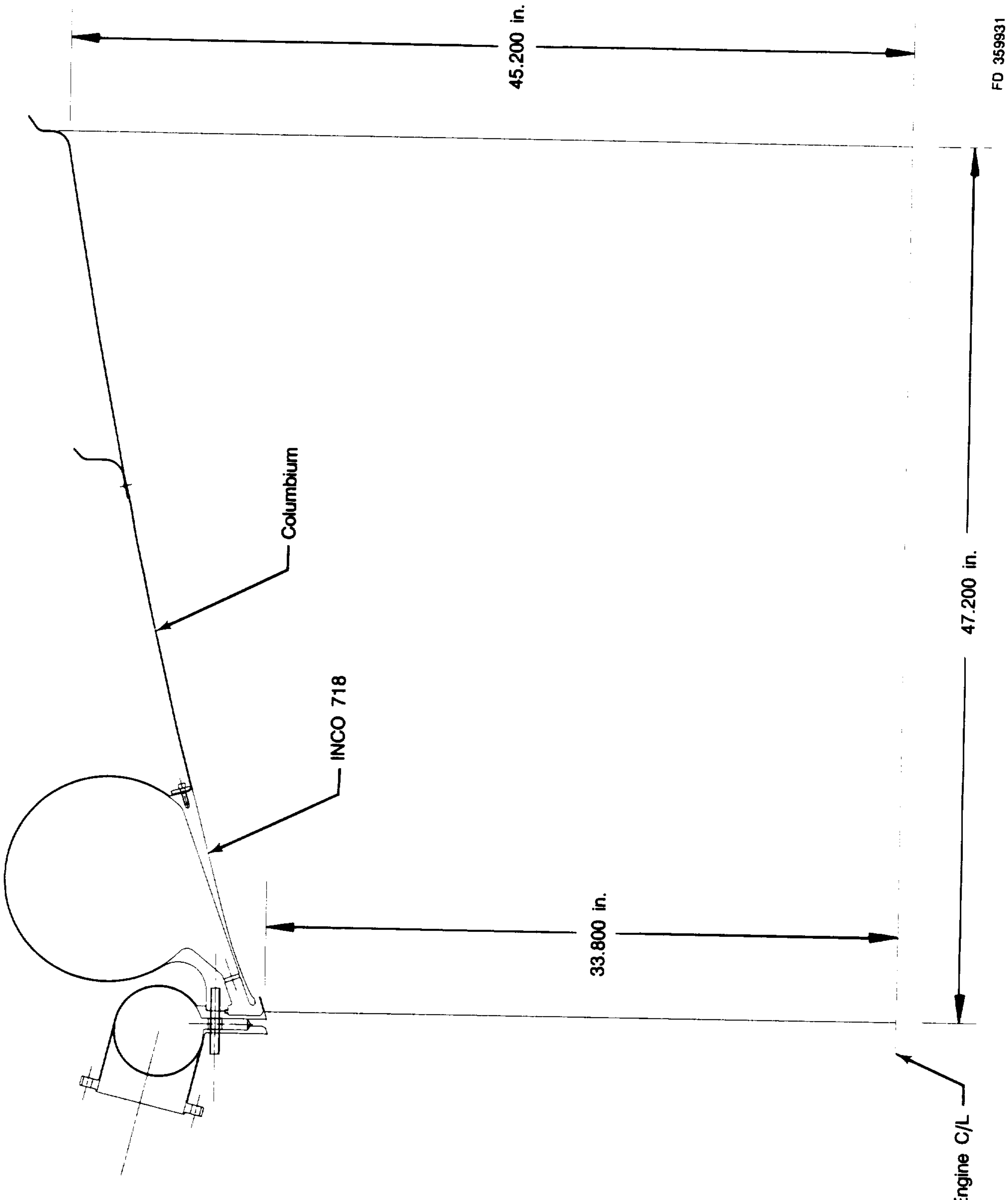


Figure 4.1.2.5-4. *STBE Unique Gas Generator Film Cooled Nozzle*

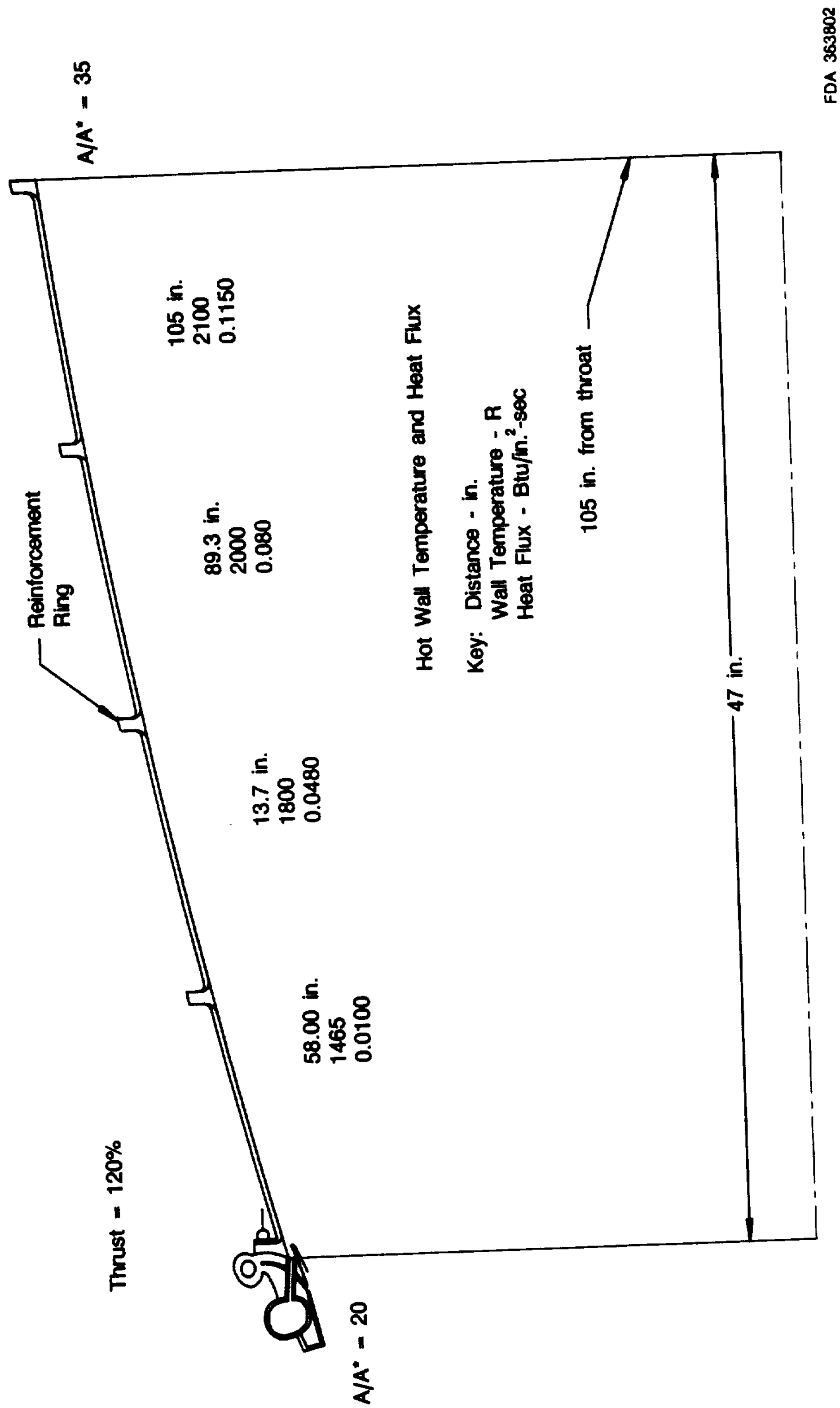


Figure 4.1.2.5-5. STBE Unique Gas Generator Film and Radiation Cooled Nozzle Heat Transfer Summary

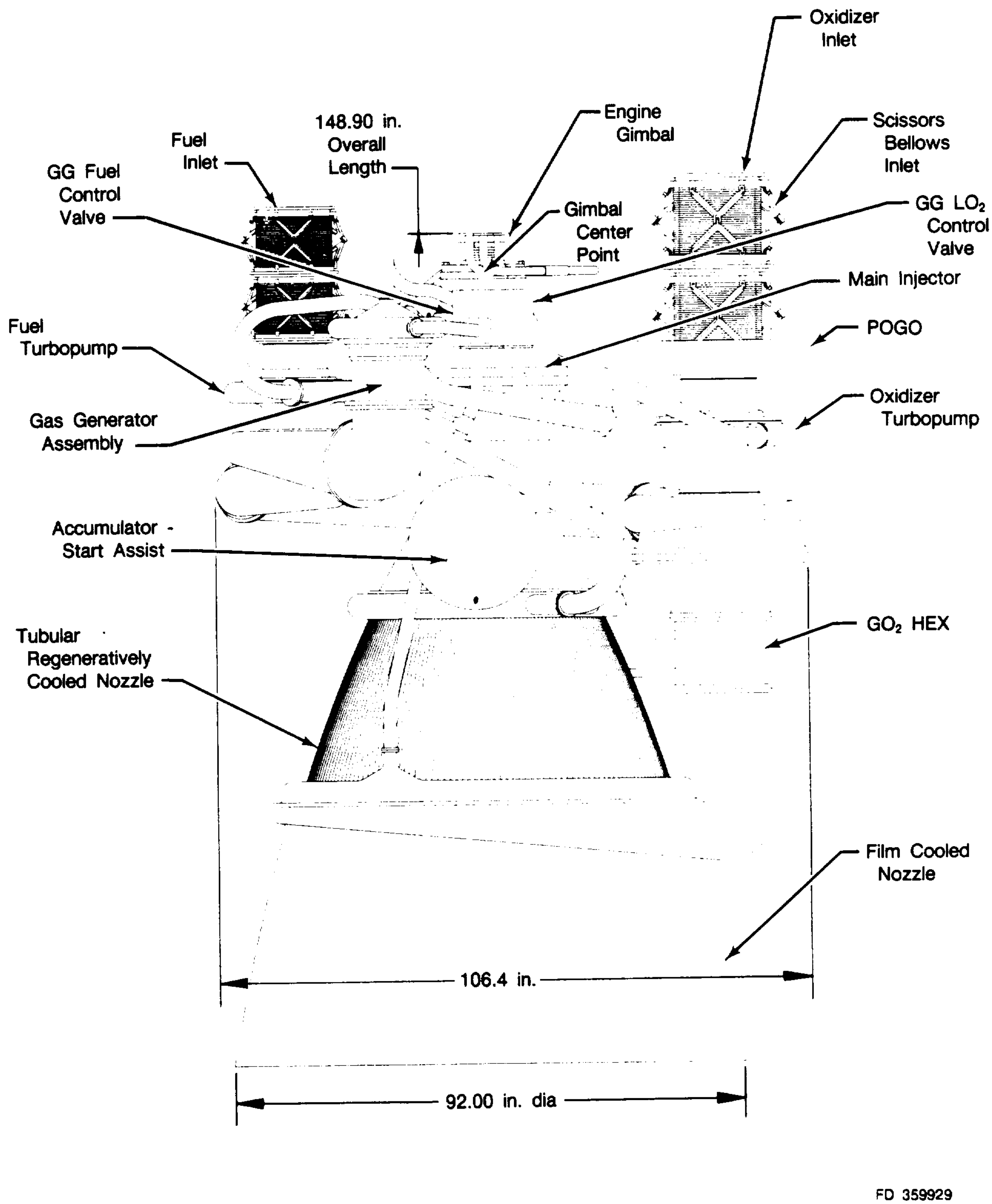


Figure 4.1.2.7-1. STBE Unique Gas Generator Engine Assembly — Side View

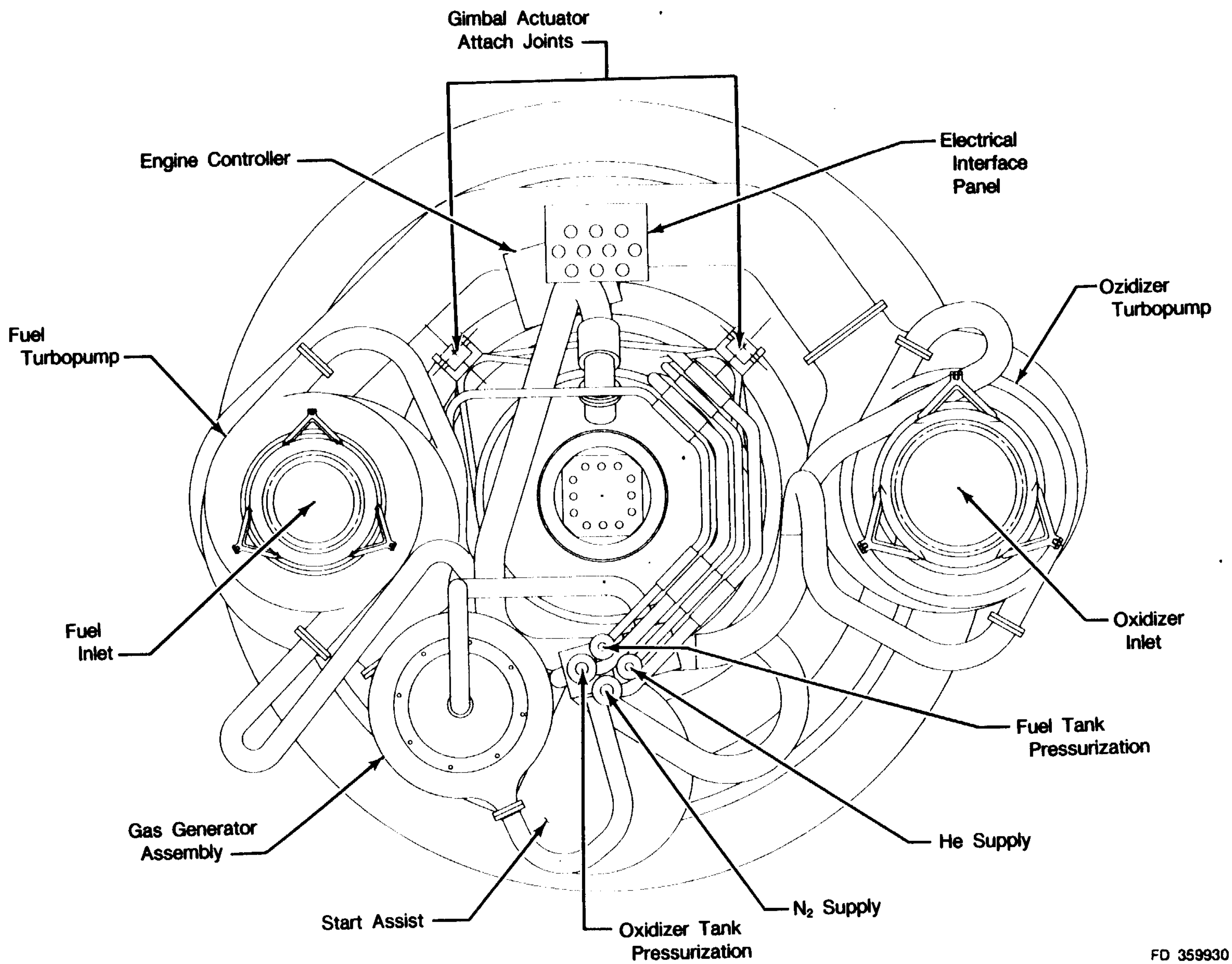


Figure 4.1.2.7-2. STBE Unique Gas Generator Engine Assembly — Top View

Turbopumps are oriented on a vertical axis and cooldown recirculation valves have been eliminated, resulting in cooldown by percolation. Engine propellant inlets accommodate engine gimballing through the use of scissor bellows mounted directly to the pump inlets. A toroidal shaped POGO accumulator has been incorporated between the LO_2 pump inlet and the scissors bellows. The engine thrust vectoring gimbal is incorporated into the main injector thrust structure. The gimbal design is based on a ball and socket feature with a central through-pin which restrains torsional movement. A teflon inpregnated fiberglass fiber woven fabric between the gimbal ball and main injector socket is used as a friction reduction medium to permit engine gimballing. Gas generator/turbine exhaust is used to provide coolant to the film cooled nozzle which is attached to the regeneratively cooled nozzle.

All pneumatic and electrical interfaces are located at the engine interface plane, similar to the SSME.

4.1.2.7.2 STBE Unique Gas Generator GO_2 Heat Exchanger

The GO_2 heat exchanger is designed to provide gaseous oxygen to the oxygen tank for tank pressurization. The GO_2 heat exchanger uses the gas generator exhaust duct flow as the heat source to vaporize the liquid oxygen as shown in Figure 4.1.2.2-1. The heat exchanger surface is provided by three Haynes 214 stainless steel tubes wrapped in parallel around the gas generator exhaust duct. The gas generator exhaust duct wall is made of beryllium copper with trip strip roughened walls to enhance the heat transfer. The tubes are packed in powdered copper to structurally isolate the tubes from the duct wall, while providing a good heat transfer medium. This heat exchanger design eliminates the possibility of accidental mixing of the oxygen and gas generator exhaust flow, thereby eliminating a category 1 failure mode.

The GO_2 heat exchanger requires three 3/8-inch diameter tubes 50-feet long, wrapped around the 12-inch duct. The tubes have 0.015-inch thick walls and are separated from one another by 0.05 inches, requiring a total duct length of 2.03 feet. Figure 4.1.2.7-3 diagramatically presents the GO_2 heat exchanger geometry.

The GO_2 heat exchanger has been thermally analyzed for the STBE engine operating point with an oxygen flow rate of 3.5 lbm/sec. The heat exchanger is designed to supply 720 R oxygen to the tank. Figure 4.1.2.7-3 also summarizes the predicted heat exchanger thermal performance.

4.1.2.7.3 Engine Performance

The STBE system performance was determined during the preliminary design using the accepted JANNAF methodology. Rigorous procedures have been established for use in calculating chamber/nozzle thrust and specific impulse. The steady-state design point computer simulation provided an initial match of components and definitions of mixture ratio, mass flow, temperature and pressure levels for the detailed performance calculations using the JANNAF methodology. Figure 4.1.2.7-4 shows a flow schematic of the JANNAF performance prediction procedure followed during this Task. Performance was estimated for both the main chamber flow and the gas generator flow, which is dumped overboard during engine operation. Table 4.1.2.7-1 lists the detailed performance estimates at the design power level (DPL) of 750,000 pounds sea level thrust while the normal power level of 625,000 pounds sea level thrust is given in Table 4.1.2.7-2. Overall engine performance was calculated by mass weighing the main chamber flow performance with the gas generator flow performance.

STBE GO_2 Heat Exchanger
Thermal and Flow Performance

Gas Generator Exhaust Duct		
Thrust - %	100	120
Temperature - R	1322	1463
Pressure - psia	186.6	261.1
Flow Rate - lbm/sec	115.4	144.9
Oxygen		
Flow Rate - lbm/sec	4.9	5.8
Inlet Temperature - R	177	179
Exit Temperature - R	720	721
Inlet Pressure - psia	2479	2987
Exit Pressure - psia	2029	2525

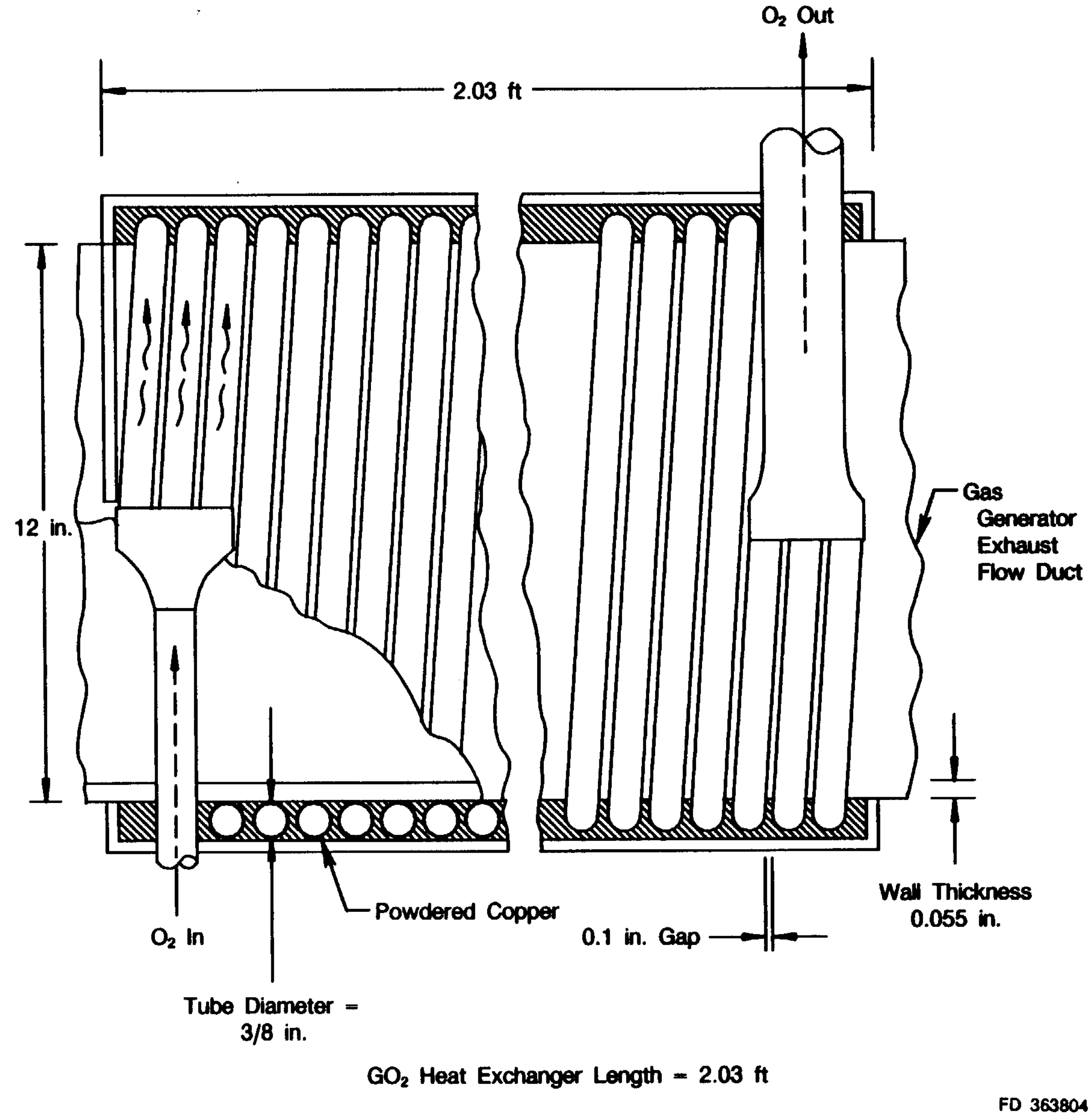


Figure 4.1.2.7-3. STBE GO_2 HEX Geometry and Performance

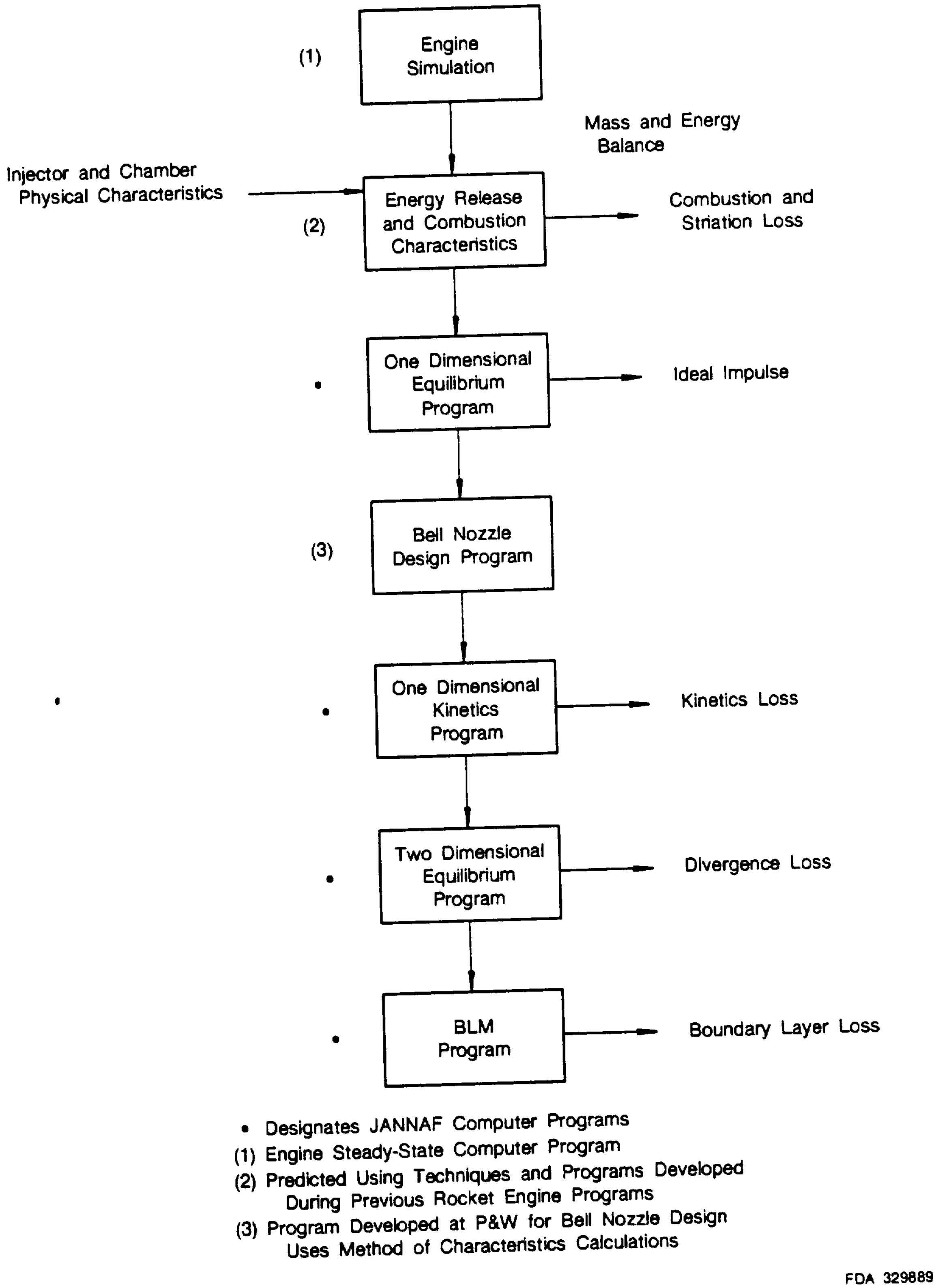


Figure 4.1.2.7-4. Performance Prediction Procedure

Table 4.1.2.7-1. Unique STBE Gas Generator Engine Performance — Design Power Level

	Main Chamber	Gas Generator
Pressure — psia	2395.9	2281.2
Mixture Ratio	3.62	0.288
Area Ratio	35	5
Ideal I_{sp} — sec	365.2	175.3
Δ I_{sp} ERE — sec	−7.3	0.0
Δ I_{sp} KIN — sec	−0.6	−4.2
Δ I_{sp} TDK — sec	−3.1	−3.8
Δ I_{sp} BLM — sec	−1.7	−0.4
Del. I_{sp} Vac — sec	352.5	166.9
Flowrate — lb/sec	2318.3	144.5
Vacuum Thrust — lb	817,275	24,118

	Overall Engine
Vacuum Thrust — lb	841,393
Vacuum Del. I_{sp} — sec	341.6
S.L. Thrust — lb	750,000
S.L. Del. I_{sp} — sec	304.5

R19691/47

Table 4.1.2.7-2. Unique STBE Gas Generator Engine Performance — Normal Power Level

	Main Chamber	Gas Generator
Pressure — psia	2054.1	1720.7
Mixture Ratio	3.5	0.223
Area Ratio	35	5
Ideal I_{sp} — sec	365.4	163.7
Δ I_{sp} ERE — sec	−7.3	0.0
Δ I_{sp} KIN — sec	−0.5	−2.9
Δ I_{sp} TDK — sec	−3.2	−3.6
Δ I_{sp} BLM — sec	−1.7	−0.4
Del. I_{sp} Vac — sec	352.7	156.8
Flowrate — lb/sec	1980.1	115.4
Vacuum Thrust — lb	698302.	18090.

	Overall Engine
Vacuum Thrust — lb	716,392
Vacuum Del. I_{sp} — sec	341.9
S.L. Thrust — lb	625,000
S.L. Del. I_{sp} — sec	298.3

R19691/47

During this study, detailed aerothermal analyses were made to predict component performance levels and these were incorporated into a steady-state computer model of the complete engine. Simplified flow schematics are presented in Figures 4.1.2.7-5 and -6 with key

operating parameters noted for each thrust level. Tables 4.1.2.7-3 and -4 define performance of the individual components and their operating environments for the STBE at DPL (120%) and at NPL (100%) respectively.

4.1.2.7.4 Engine Costs

This section summarizes cost estimates for the 750K SL thrust, 2396 psia chamber pressure, Unique STBE Gas Generator cycle. Table 4.1.2.7-5 summarizes significant costs for the engine.

The DDT&E Cost includes all of the functions required to design, develop, test and evaluate the engine system. All of the DDT&E functions shown in the ALS engine WBS (see Volume III) have been included. Development Cost is based on a 90-month phase C/D program with 960 engine firings for the Unique STBE Gas Generator. Sufficient accountable firings have been included in the program to demonstrate 0.99 engine reliability with one failure.

The engine Theoretical First Unit (TFU) production cost includes all the recurring operational production cost elements specified in the ALS engine WBS. It includes manufacturing and acceptance of the Integrated Engine System, System Engineering and Integration, Program Management, Facilities Maintenance and Tooling Maintenance. The TFU estimate is based on a lot size of 100 and a 90-percent learning curve.

The Operations Cost per launch per engine includes all costs associated with the operational flight program as described in the ALS engine WBS. It includes Program Management, System Engineering and Integration, Facilities Maintenance, Operation and Support, and Training. The Operations Cost is based on a flight rate of 10 missions per year and it is the estimated cost that will be achieved after 100 total missions have been flown.

4.1.3 Common STBE Gas Generator Cycle Engine

4.1.3.1 Engine Design Evolution

The common STBE/STME Gas Generator Cycle Engine design has evolved from a 388,000-pound (388K) sea level design thrust, very common engine to a higher thrust with considerable part commonality but minimal performance penalty to the STME. The common engine concepts were based upon the following guidelines during conceptual design studies:

- Use unique STME hardware wherever possible for both the STBE and STME engines

- Where unique STME engine hardware cannot be used for both engines, design the most common piece of hardware, while minimizing performance debit to the STME engine, i.e., main injector

- Where a common piece of hardware could not be used, (such as the main combustion chamber), design a new part for the booster engine application, and use the unique STME design for the main engine application.

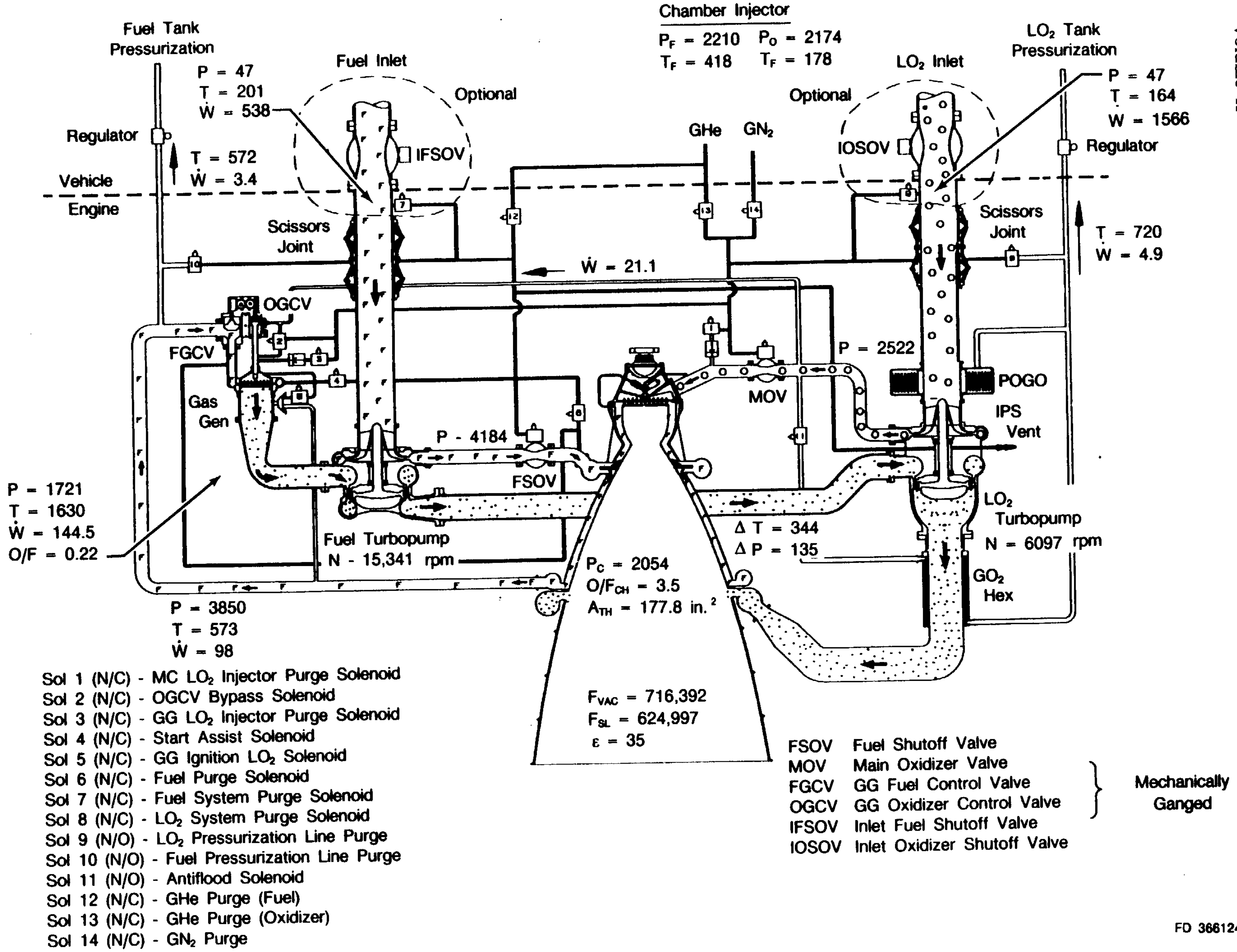
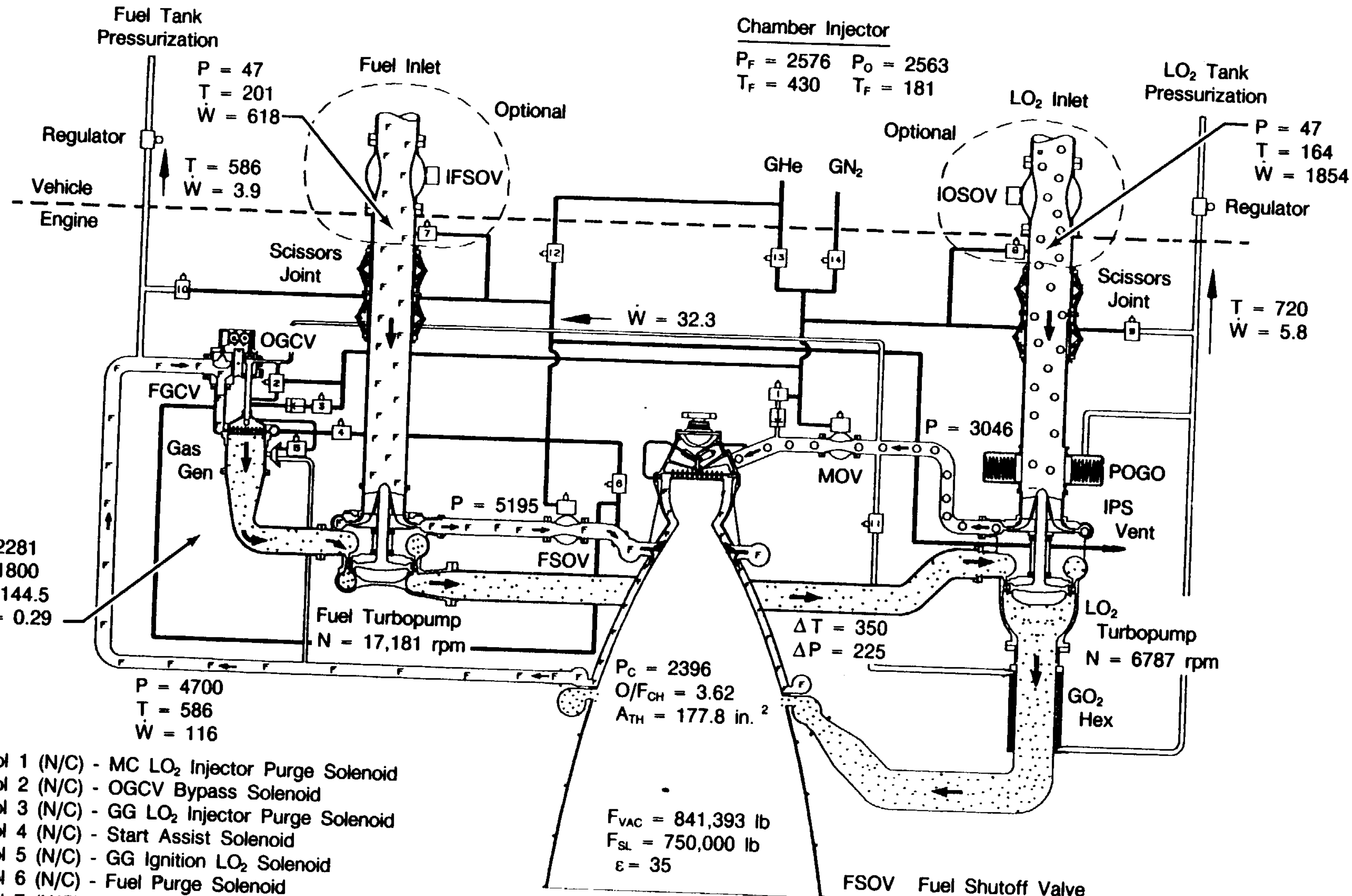


Figure 4.1.2.7-5. STBE Unique Gas Generator Cycle Engine Operating Characteristics at Normal Power Level



Sol 1 (N/C) - MC LO_2 Injector Purge Solenoid
Sol 2 (N/C) - OGCV Bypass Solenoid
Sol 3 (N/C) - GG LO_2 Injector Purge Solenoid
Sol 4 (N/C) - Start Assist Solenoid
Sol 5 (N/C) - GG Ignition LO_2 Solenoid
Sol 6 (N/C) - Fuel Purge Solenoid
Sol 7 (N/C) - Fuel System Purge Solenoid
Sol 8 (N/C) - LO_2 System Purge Solenoid
Sol 9 (N/O) - LO_2 Pressurization Line Purge
Sol 10 (N/O) - Fuel Pressurization Line Purge
Sol 11 (N/O) - Antiflood Solenoid
Sol 12 (N/C) - GHe Purge (Fuel)
Sol 13 (N/C) - GHe Purge (Oxidizer)
Sol 14 (N/C) - GN_2 Purge

FSOV	Fuel Shutoff Valve	
MOV	Main Oxidizer Valve	
FGCV	GG Fuel Control Valve	Mechanically Ganged
OGCV	GG Oxidizer Control Valve	Mechanically Ganged
IFSOV	Inlet Fuel Shutoff Valve	
IOSOV	Inlet Oxidizer Shutoff Valve	

FD 363211

Figure 4.1.2.7-6. STBE Unique Gas Generator Cycle Engine Operating Characteristics at Design Power Level

Table 4.1.2.7-3. Unique STBE Gas Generator Engine Performance — Design Power

Unique STBE Gas Generator Engine Performance — Design Power Level

PRATT & WHITNEY
GAS GENERATOR CYCLE OFF-DESIGN DECK
STBE ENGINE STUDY

ENGINE PERFORMANCE

VACUUM THRUST	841393.
SEA LEVEL THRUST	749998.
VACUUM IMPULSE	341.64
SEA LEVEL IMPULSE	304.53
TOTAL ENGINE INLET FLOW RATE	2472.5
OVERALL ENGINE MIXTURE RATIO	3.00

CHAMBER PERFORMANCE

PRESSURE	2395.9
TEMPERATURE	6636.5
THRUST	817275.
IMPULSE	352.53
FLOW RATE	2318.3
THROAT AREA	177.76
NOZZLE AREA RATIO	35.
MIXTURE RATIO	3.62
NOZZLE EFFICIENCY	0.980
CSTAR EFFICIENCY	0.980

ENGINE HEAT TRANSFER

CHAMBER COOLANT DP	2331.
CHAMBER COOLANT DT	194
CHAMBER Q	83460.
NOZZLE COOLANT DP	225.
NOZZLE COOLANT DT	350.
NOZZLE Q	33113.

GAS GENERATOR PERFORMANCE

PRESSURE	2281.2
TEMPERATURE	1800.0
THRUST	24118.
IMPULSE	166.92
FLOW RATE	144.5
MIXTURE RATIO	0.288
NOZZLE EFFICIENCY	0.980
GAS CONSTANT	95.8
GAMMA	1.159
NOZZLE AREA	59.2

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	201.0	618.1	122.8	26.41
MAIN PUMP EXIT	5195.2	234.8	618.1	173.4	26.75
CHAM/COOL INLET	4933.5	236.4	502.1	173.4	26.61
CHAM/COOL EXIT	2602.5	430.4	502.1	339.7	14.35
CH INJ INLET	2576.3	430.0	502.1	339.7	14.29
NOZ/COOL INLET	4924.9	236.4	116.1	173.4	26.60
NOZ/COOL EXIT	4700.1	586.0	116.1	458.7	12.31
TANK PRESS OUT	4622.1	585.5	3.9	458.7	12.19
TANK PRESS IN	47.0	432.7	3.9	458.7	0.16
FGCV INLET	4622.1	585.5	112.2	458.7	12.19
FGCV EXIT	2443.6	553.8	112.2	458.7	7.86
GG INJ INLET	2409.9	553.0	112.2	458.7	7.78

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1854.4	62.0	70.98
MAIN PUMP EXIT	3046.3	178.6	1854.4	72.3	71.53
TANK PRESS OUT	2986.6	178.8	5.8	72.3	71.44
TANK PRESS IN	47.0	720.0	5.8	275.4	0.22
MOV INLET	2986.6	178.8	1816.3	72.3	71.44
MOV EXIT	2615.8	180.3	1816.3	72.3	70.87
CH INJ INLET	2563.2	180.6	1816.3	72.3	70.79
OGCV INLET	2914.2	179.1	32.3	72.3	71.33
OGCV EXIT	2531.2	180.7	32.3	72.3	70.74
GG INJ INLET	2471.4	180.9	32.3	72.3	70.65

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
FUEL TURB INLET	2235.5	1800.0	144.5
FUEL TURB EXIT	648.2	1592.5	144.5
LOX TURB INLET	615.7	1589.2	144.5
LOX TURB EXIT	261.1	1462.9	144.5
NOZZLE INLET PRES	248.0		

PRATT & WHITNEY
GAS GENERATOR CYCLE OFF-DESIGN DECK
STBE ENGINE STUDY

TURBOMACHINERY PERFORMANCE DATA

* FUEL TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.858	0.849
HORSEPOWER	25228.	19088.
SPEED (RPM)	17181.	17181.
S SPEED	33.4	52.2
S DIAMETER	1.87	1.41
MEAN DIAMETER (IN)	13.80	13.80
VEL.RATIO (ACTUAL)	0.42	0.48
MAX TIP SPEED	1092.	1129.
BLADE HEIGHT	0.75	1.25
AN SQUARED	96.0	160.0
EFFECTIVE AREA	7.39	14.01
PRES.RATIO (T/T)	1.97	1.75

* FUEL PUMP *

	STAGE ONE
EFFICIENCY	0.705
HORSEPOWER	44316.
SPEED (RPM)	17181.
NPSP (PSI)	32.6
SS SPEED	44574.
S SPEED	813.
HEAD (FT)	27786.
DIAMETER (IN)	16.80
TIP SPEED (FT/SEC)	1260.
VOL FLOW	10504.
HEAD COEF	0.5574
FLOW COEF	0.1284

* LOX TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.844	0.830
HORSEPOWER	13441.	13437.
SPEED (RPM)	6787.	6787.
S SPEED	33.0	39.7
S DIAMETER	1.81	1.56
MEAN DIAMETER (IN)	24.50	24.50
VEL.RATIO (ACTUAL)	0.40	0.40
MAX TIP SPEED	771.	808.
BLADE HEIGHT	1.50	2.75
AN SQUARED	53.2	97.5
EFFECTIVE AREA	25.53	37.38
PRES.RATIO (T/T)	1.51	1.56

* LOX PUMP *

EFFICIENCY	0.759
HORSEPOWER	26878.
SPEED (RPM)	6787.
NPSP (PSI)	30.9
SS SPEED	35728.
S SPEED	1067.
HEAD (FT)	6049.
DIAMETER (IN)	20.40
TIP SPEED (FT/SEC)	605.
VOL FLOW	11727.
HEAD COEF	0.5317
FLOW COEF	0.1441

* VALVE DATA *

STATION	DELP	AREA	FLOW
FUEL GG VALVE	2178.5	1.387	112.2
MAIN OXID VALVE	370.9	16.69	1816.3
LOX GG VALVE	383.0	0.292	32.3

* INJECTOR DATA *

STATION	DELP	AREA	FLOW
FUEL GG INJ	128.6	5.931	112.2
FUEL CH INJ	180.4	19.10	502.1
LOX GG INJ	190.2	0.417	32.3
LOX CH INJ	167.3	24.97	1816.3

Table 4.1.2.7-4. Unique STBE Gas Generator Engine Performance — Normal Power

Unique STBE Gas Generator Engine Performance — Normal Power Level

PRATT & WHITNEY
GAS GENERATOR CYCLE OFF-DESIGN DECK
STBE ENGINE STUDY

ENGINE PERFORMANCE

VACUUM THRUST	716392.
SEA LEVEL THRUST	624997.
VACUUM IMPULSE	341.88
SEA LEVEL IMPULSE	298.26
TOTAL ENGINE INLET FLOW RATE	2103.8
OVERALL ENGINE MIXTURE RATIO	2.91

CHAMBER PERFORMANCE

PRESSURE	2054.1
TEMPERATURE	6586.6
THRUST	698302.
IMPULSE	352.66
FLOW RATE	1980.1
THROAT AREA	177.76
NOZZLE AREA RATIO	35.
MIXTURE RATIO	3.50
NOZZLE EFFICIENCY	0.980
CSTAR EFFICIENCY	0.980

ENGINE HEAT TRANSFER

CHAMBER COOLANT DP	1754.
CHAMBER COOLANT DT	189.
CHAMBER Q	74296.
NOZZLE COOLANT DP	135.
NOZZLE COOLANT DT	343.
NOZZLE Q	28416.

GAS GENERATOR PERFORMANCE

PRESSURE	1720.7
TEMPERATURE	1630.3
THRUST	18090.
IMPULSE	156.80
FLOW RATE	115.4
MIXTURE RATIO	0.223
NOZZLE EFFICIENCY	0.980
GAS CONSTANT	94.4
GAMMA	1.159
NOZZLE AREA	59.2

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	201.0	537.6	122.8	26.41
MAIN PUMP EXIT	4184.6	228.2	537.6	163.6	26.68
CHAM/COOL INLET	3985.1	229.3	439.9	163.6	26.57
CHAM/COOL EXIT	2231.1	417.9	439.9	332.5	13.99
CH INJ INLET	2210.4	417.5	439.9	332.5	13.94
NOZ/COOL INLET	3984.9	229.3	97.7	163.6	26.57
NOZ/COOL EXIT	3850.3	572.8	97.7	454.3	11.06
TANK PRESS OUT	3788.7	572.1	3.4	454.3	10.95
TANK PRESS IN	47.0	424.0	3.4	454.3	0.17
FGCV INLET	3788.7	572.1	94.3	454.3	10.95
FGCV EXIT	1865.2	530.9	94.3	454.3	6.42
GG INJ INLET	1836.0	529.8	94.3	454.3	6.34

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1566.2	62.0	70.98
MAIN PUMP EXIT	2521.7	176.4	1566.2	70.5	71.41
TANK PRESS OUT	2479.1	176.5	4.9	70.5	71.35
TANK PRESS IN	47.0	720.0	4.9	275.4	0.22
MOV INLET	2479.1	176.5	1540.2	70.5	71.35
MOV EXIT	2212.1	177.5	1540.2	70.5	70.94
CH INJ INLET	2174.3	177.7	1540.2	70.5	70.88
OGCV INLET	2448.2	176.6	21.1	70.5	71.30
OGCV EXIT	1827.5	179.0	21.1	70.5	70.33
GG INJ INLET	1801.9	179.0	21.1	70.5	70.29

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
FUEL TURB INLET	1686.3	1630.3	115.4
FUEL TURB EXIT	488.7	1441.1	115.4
LOX TURB INLET	464.1	1438.6	115.4
LOX TURB EXIT	196.6	1322.2	115.4
NOZZLE INLET PRES	186.8		

PRATT & WHITNEY
GAS GENERATOR CYCLE OFF-DESIGN DECK
STBE ENGINE STUDY

TURBOMACHINERY PERFORMANCE DATA

* FUEL TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.841	0.835
HORSEPOWER	17598.	13423.
SPEED (RPM)	15341.	15341.
S SPEED	31.6	49.3
S DIAMETER	1.87	1.41
MEAN DIAMETER (IN)	13.80	13.80
VEL.RATIO (ACTUAL)	0.40	0.46
MAX TIP SPEED	975.	1008.
BLADE HEIGHT	0.75	1.25
AN SQUARED	76.5	127.5
EFFECTIVE AREA	7.39	14.01
PRES.RATIO (T/T)	1.97	1.75

* FUEL PUMP *

	STAGE ONE
EFFICIENCY	0.705
HORSEPOWER	31022.
SPEED (RPM)	15341.
NPSP (PSI)	32.6
SS SPEED	37118.
S SPEED	797.
HEAD (FT)	22380.
DIAMETER (IN)	16.80
TIP SPEED (FT/SEC)	1125.
VOL FLOW	9136.
HEAD COEF	0.5631
FLOW COEF	0.1251

* LOX TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.830	0.817
HORSEPOWER	9421.	9439.
SPEED (RPM)	6097.	6097.
S SPEED	31.4	37.7
S DIAMETER	1.81	1.56
MEAN DIAMETER (IN)	24.50	24.50
VEL.RATIO (ACTUAL)	0.38	0.38
MAX TIP SPEED	692.	726.
BLADE HEIGHT	1.50	2.75
AN SQUARED	42.9	78.7
EFFECTIVE AREA	25.53	37.38
PRES.RATIO (T/T)	1.51	1.56

* LOX PUMP *

EFFICIENCY	0.755
HORSEPOWER	18861.
SPEED (RPM)	6097.
NPSP (PSI)	30.9
SS SPEED	29496.
S SPEED	1018.
HEAD (FT)	4997.
DIAMETER (IN)	20.40
TIP SPEED (FT/SEC)	543.
VOL FLOW	9905.
HEAD COEF	0.5444
FLOW COEF	0.1355

* VALVE DATA *

STATION	DELP	AREA	FLOW
FUEL GG VALVE	1923.5	1.387	94.3
MAIN OXID VALVE	267.0	16.69	1540.2
LOX GG VALVE	620.8	0.150	21.1

* INJECTOR DATA *

STATION	DELP	AREA	FLOW
FUEL GG INJ	115.2	5.931	94.3
FUEL CH INJ	156.3	19.10	439.9
LOX GG INJ	81.2	0.417	21.1
LOX CH INJ	120.2	24.97	1540.2

Table 4.1.2.7-5. Unique STBE Gas Generator Costs

Total Development Cost (DDT&E), M$1329*
Production Cost (TFU), M$11.2
Operations Cost/Launch/Engine, M$0.150**
Constant FY87$

*Applies to developing a stand-alone booster engine configuration.
**Based on the 100th mission, 10 missions per year, and seven boosters per vehicle.

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Four separate engine designs resulted from this commonality philosophy in the STME and STBE programs:

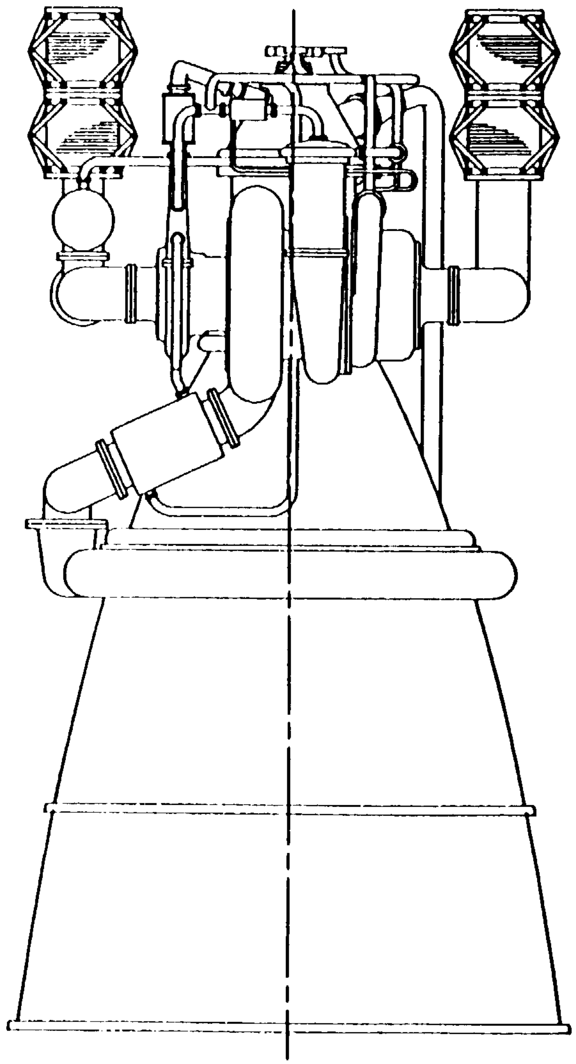
1. Unique STME
2. Common STME (similar to the unique STME with slight performance, cost, and weight penalties)
3. Unique STBE
4. Common STBE (significant performance penalty when compared to the unique STBE).

The first common engine design, in which a common main combustion chamber was used, resulted in a low-thrust booster engine design. The engine external assembly and characteristics are shown in Figure 4.1.3.1-1, for operation with both LO_2/H_2 and LO_2/CH_4 as propellants.

This low-thrust level in the STBE that resulted from a most common STME/STBE engine proved to be unacceptable for use as an ALS booster engine. Therefore, design changes required to produce a 750K sea level thrust STBE engine resulted in a new design (but common) main combustion chamber and major pump housings. The engine assembly design and major characteristics are shown in Figure 4.1.3.1-2. Due to the higher fuel system pressures in the STBE cycle, the common chamber, controls, pump housings, and large ducting lines imposed a large weight penalty on the STME. The common STME thrust-to-weight ratio was approximately 56.5:1, while the unique STME thrust-to-weight ratio was 85:1. The results of this study prompted P&W to back off on the second guideline, engine commonality, and design separate, unique main combustion chambers, major turbopump housings, and large ducting lines and controls for each engine. This engine assembly and major characteristics are shown in Figure 4.1.3.1-3. This engine design was further refined to minimize performance, cost, and weight penalties to the STME, while maximizing part commonality between the two common engines and maintaining STBE thrust at an acceptable level of 635K sea level thrust. A comparative summary of the major engine components for the Unique 580K STME design, the Common 580K STME design, and the 635K STBE engine is presented in Table 4.1.3.1-1.

4.1.3.2 Engine Cycle

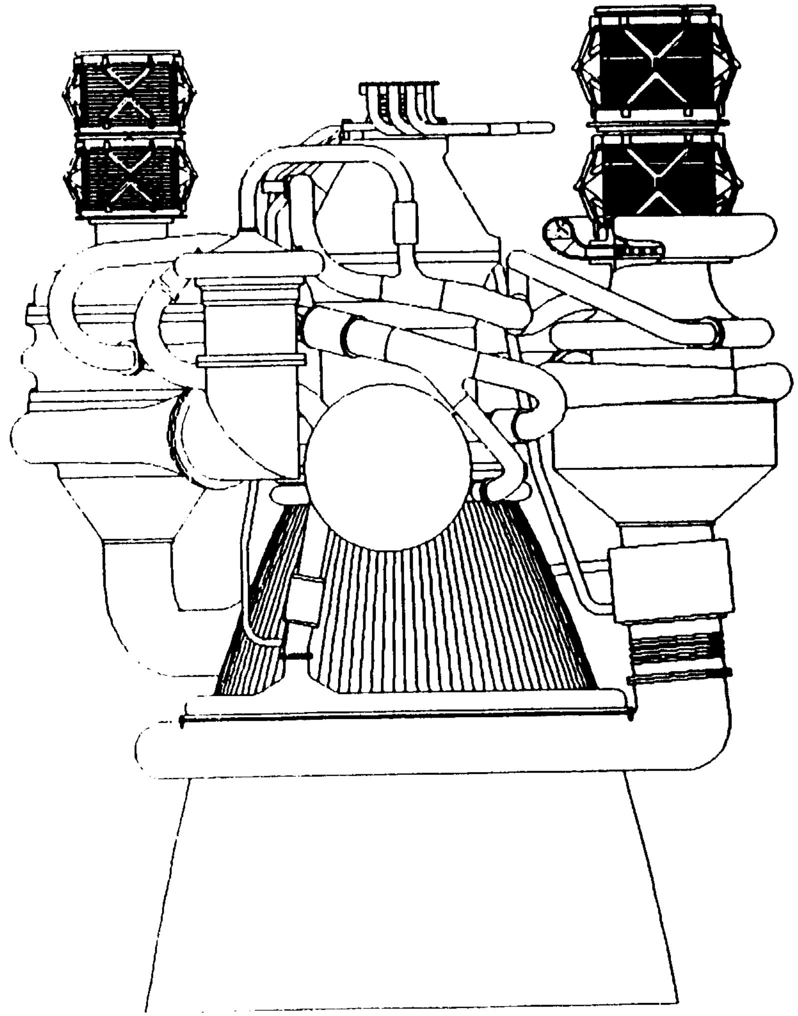
The STME/STBE common gas generator configuration, studied during the Phase A' contract, uses liquid oxygen and liquid hydrogen as propellants for the STME engine; while liquid oxygen and liquid methane is the fuel for the STBE engine. This engine operates at a main chamber pressure of 2250 psia at the Design Power Level (DPL) of 580K lbf vacuum thrust for the STME; and 635K lbf sea level thrust for the STBE. The STME engine has a fixed nozzle with an area ratio of 62:1 that delivers 440.0 seconds of vacuum specific impulse at DPL. The STBE engine uses an STME nozzle that is truncated at an area ratio of 35:1 and delivers 295.4 seconds of sea level specific impulse. Figure 4.1.3.1-3 presents selected engine characteristics at the design power level for the STME/STBE Common engine.



Propellants	LO_2/H_2	LO_2/CH_4
Mixture Ratio	6.0	3.0
Chamber Pressure	2678 psia	2250 psia
Thrust - Vacuum	435,000 lb	349,700 lb
- Sea Level	340,700 lb	312,700 lb
Specific Impulse - Vacuum	448.1 sec	344.6 sec
- Sea Level	351.0 sec	308.0 sec
Nozzle Area Ratio	80	35
Diameter	96 in.	64 in.
Length	168 in.	92 in.
Weight	4731 lb	TBD lb

FD 359981

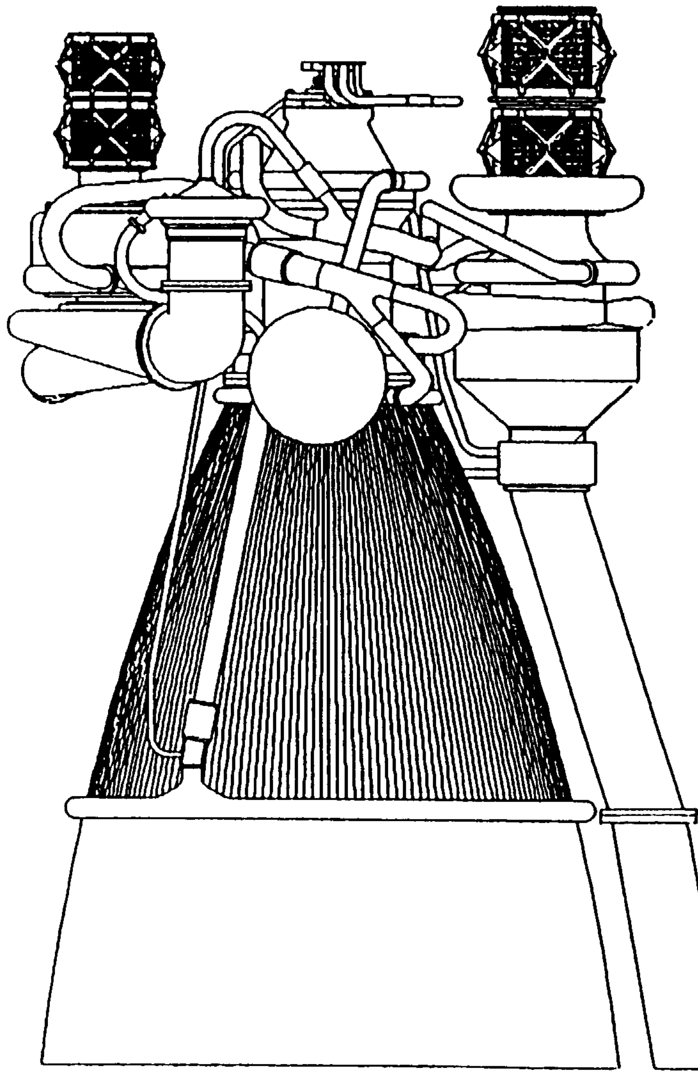
Figure 4.1.3.1-1. STBE Gas Generator Common Engine at Normal Operating Conditions



Propellants	LO_2/H_2	LO_2/CH_4
Mixture Ratio	6.0	2.70
Chamber Pressure	2250 psia	3159 psia
Thrust - Vacuum	580,000 lb	816,225 lb
- Sea Level	462,750 lb	750,024 lb
Specific Impulse - Vacuum	436.5 sec	337.3 sec
- Sea Level	348.2 sec	309.9 sec
Nozzle Area Ratio	62	35
Diameter	104 in.	76 in.
Length	174 in.	133 in.
Weight	10,254 lb	10,766 lb

FD 359982

Figure 4.1.3.1-2. STBE Gas Generator Common Engine at Design Operating Conditions



Propellants	LO_2/H_2	LO_2/CH_4
Mixture Ratio	6.0	3.0
Chamber Pressure	2250 psia	2250 psia
Thrust - Vacuum	580,000 lb	719,035 lb
- Sea Level	460,921 lb	635,008 lb
Specific Impulse - Vacuum	440.1 sec	334.5 sec
- Sea Level	349.7 sec	295.4 sec
Nozzle Area Ratio	62	35
Diameter	102 in.	91 in.
Length	179 in.	136 in.
Weight	TBD lb	TBD lb

FD 359983

Figure 4.1.3.1-3. STBE Gas Generator Common Engine at Design Operating Conditions

Table 4.1.3.1-1. STME/STBE Common Engine and STME $H_2/0_2$ Unique Engine Comparison — Common Hardware Evaluation

Component	Unique STME	Common GG H_2 Engine	Common GG CH_4 Engine
Fuel System			
Pump	Unique	New Design	Same as H_2
Vent Valve	Unique	Same as CH_4	New Design
Shutoff Valve	Unique	Same as CH_4	New Design
Coolant Valve	Unique	Same as CH_4	New Design
GG Control Valve	Unique	Same as CH_4	New Design
Gas Generator	Unique	New Design	Same as H_2
Turbine	Unique	New Design	Reblade From H_2, Same Housings, etc.
Oxidizer System			
POGO Suppressor	Common	Same as STME	Same as STME
Vent Valve	Unique	Same as CH_4	New Design
Main Valve	Unique	Same as CH_4	New Design
Heat Exchanger	Common	Same as STME	Same as STME
Turbine	Unique	New Design	Reblade From H_2, Same Housings, etc.
Chamber and Injector			
Injector	Unique	New Design	Same as H_2
Regeneratively Cooled Nozzle	Unique	New Design	Same as H_2
Film Cooled Nozzle	Unique	New Design	No Additional Nozzle
Igniter	Common	Same as STME	Same as STME, Modified Flow Control Orifices
Combustor	Unique	New Design	New Design With Acoustic Liner
Controls			
Instrumentation	Common	Same as STME	Same as STME
Engine Controller	Common	Same as STME	Same as STME Software Change
Engine Assembly			
Engine Ducting	50% Common	50% Same as STME	Same as H_2
Vehicle Interfaces	Common	Same as STME	Same as STME
Gimbal	Common	Same as STME	Same as STME

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The benefit of engine commonality is the reduction of manufacturing costs. The common hardware between the STME/STBE Common gas generator engines are as follows: pumps, turbines, gas generator, combustor, nozzle, igniter, injectors, controls, GO_2 heat exchanger, LO_2 POGO suppressor, LO_2 vent and main valves. However, some modifications had to be made and are: restaggering of STBE turbine blades; truncation of the STBE nozzle at an area ratio of 35:1; change fuel orifices in the STBE injector; and some software changes in the engine controller.

4.1.3.2.1 Flow Path Description

A simplified flow schematic for the STME/STBE common engine is presented in Figure 4.1.3.2-1, showing the major flowpaths and components.

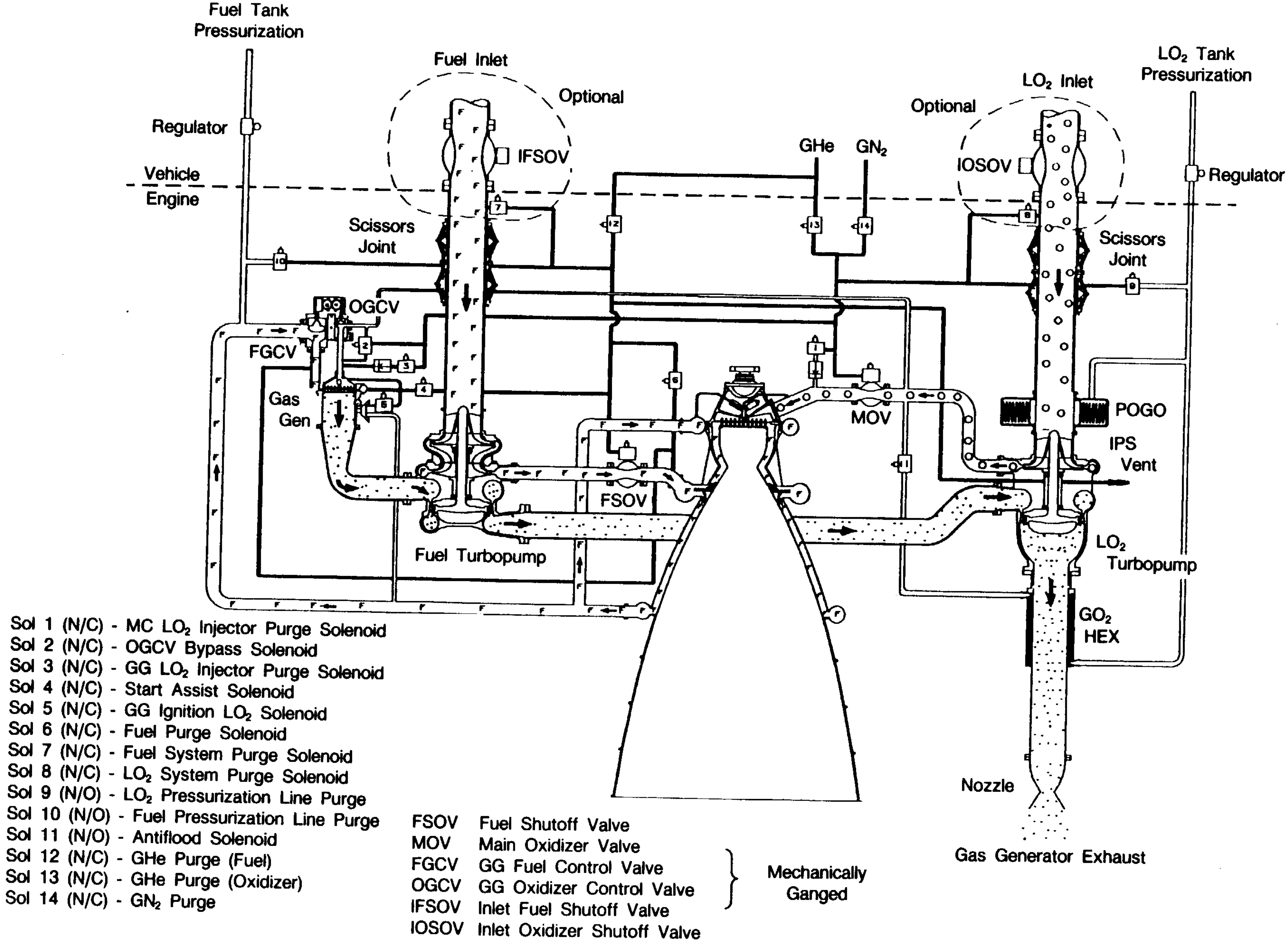

Figure 4.1.3.2-1. Simplified Flow Schematic for STME/STBE Common Gas Generator Cycle Engine

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure oxidizer pump. The fuel enters the engine at a NPSH level, again supplied by the vehicle, sufficient for the high-speed high-pressure fuel pump; thus boost pumps are not required for this system.

At the Design Power Level for the STME (STBE), the fuel pump operates at 21,660 rpm (10,478 rpm) to provide a fuel pressure level of 3456 psia (4710 psia) required by the cycle. From the pump exit, the fuel flows through the fuel shutoff valve to a split manifold at the inlet of the coolant passages. From the split manifold, 70.5 percent (45.5 percent) of the fuel is used to regeneratively cool the milled-channel copper alloy main chamber from an area ratio of 5.86:1 back to the injector face. Then it is routed directly into the injector manifold and then the main combustion chamber. The remaining fuel flow is used to cool the tubular stainless steel nozzle from an area ratio of 35:1. Subsequently, the nozzle cooling flow splits where 38.7 percent (37.5 percent) is supplied to the gas generator and the rest is routed on to the main thrust chamber. The fuel supplied to the gas generator control valve is injected into the gas generator to combust with some of the oxygen to provide power for the high pressure turbomachinery.

The high-pressure oxidizer pump operates at 6435 rpm (7500 rpm) to provide the oxygen pressure level of 2784 psia (3336 psia) required by the cycle at the Design Power Level. From the pump exit, approximately 98 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

The gas generator provides 1175 psia (2400 psia), 1800 R gas to drive the high-pressure propellant pumps. The hot gas is initially expanded through the fuel turbine and is subsequently routed to a second turbine which powers the oxygen pump. The turbine exhaust gas is then expanded through an area ratio of 5:1 to atmospheric pressure; thus providing additional thrust to the overall engine output.

Some key design conditions for the turbopumps and chamber are listed in Table 4.1.3.2-1.

Table 4.1.3.2-1. *Common STME/STBE GG Design Conditions*

	STME	STBE
HPFTP		
Pressure — psia	3456	4710
Maximum Speed — rpm	21660	10478
Turbine Temperature — R	1800	1800
Pump Stages	2	2
Turbine Stages	2	2
HPOTP		
Pressure — psia	2784	3336
Maximum Speed — rpm	6435	7500
Turbine Temperature — R	1317	1624
Pump Stages	1	1
Turbine Stages	2	2
Heat Transfer		
Chamber Pressure — psia	2250	2250
GG Pressure — psia	1175	2400
Chamber Heat Pickup — btu/sec	78,048	58,849
Chamber Coolant Flow — lbm/sec	133.1	245.5
Nozzle Heat Pickup — btu/sec	51,625	68,215
Nozzle Coolant Flow — lbm/sec	55.8	294.4

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4.1.3.2.2 Engine Operation

The engine will be preconditioned using liquid flow from the tanks to soak the turbopumps until they are sufficiently cooled. The inlet valves will be opened, allowing liquid from the tanks

to flow down to the turbopumps and letting any vapors to percolate back up to the tank to be vented.

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor during the oxygen phase transition from gas to liquid. The transition occurs prior to fuel injection and the fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the gas generator LO_2 injector is primed prior to opening the fuel shutoff valve to ensure liquid oxygen flow, eliminating turbine temperature spikes due to oxygen phase change. A helium spin assist is also utilized to initiate turbopump rotation before the fuel is introduced into the gas generator. During the start and shutdown, a small helium purge is used in the gas generator injector and main chamber injector to eliminate the danger of hot gas flow reversals during transient operations. Gas generator and main chamber ignition will be accomplished with dual electrical spark-excited, oxygen/fuel torch igniters.

Main stage engine operation is open-loop controlled. The fuel gas generator control valve (FGCV), the oxygen gas generator control valve (OGCV), and the main oxidizer valve (MOV), shown in Figure 4.1.3.2-1 are used to set the engine thrust and mixture ratio. Thrust and main chamber mixture ratios are set on the ground by trimming the MOV and OGCV, respectively. The gas generator mixture ratio is set using the FGCV. All valves are operated by hydraulic actuators.

Engine acceleration is accomplished by a time-based scheduling of the valves to the commanded starting level (20 percent power level). The acceleration to full thrust is also accomplished with open-loop valve schedules. Engine shutdown is accomplished using a time-based scheduling of the propellant valves. The OGCV is closed first to power down the turbopumps, then the MOV closes, followed by shutting off the fuel system.

In addition to a normal operational mode, the engine system is capable of shutdown resulting from detected problems or LO_2 starvation at the end of the burn duration.

4.1.3.3 Turbomachinery

4.1.3.3.1 Oxidizer Turbopump

The mechanical description of the features of this turbopump are the same as the STBE Derivative Gas Generator Oxidizer Turbopump. Figure 4.1.3.3-1 shows the oxidizer turbopump and its major components.

The P&W Advanced Launch Systems (ALS) program is designed to produce reliable, low-cost rocket engine turbopumps. Pratt & Whitney uses proven design criteria and analytical methods in the design of rotordynamic operation for jet engine rotors and rocket engine turbopumps. Each Common Gas Generator Oxygen Turbopump (CGGOT) and Common Gas Generator Fuel Turbopump (CGGFT) design incorporates configuration changes which result in stiffer rotors, bearings, and rotor support structures with the addition of roughened stator damper seals. For optimum rotordynamics, each rotor is supported by strategically located, stiff, durable bearings. These changes result in a significant improvement to the first fundamental bending mode of the rotor, moving it well beyond the maximum operating design speed. This, in addition to an improved rotor balance procedure, results in an effective low speed balance of the rotor for low synchronous response. Rotor stability in the CGGOT and the CGGFT have been

improved by designing the turbopumps to operate below the first vibrational mode of the rotor. Increased stability margin in each turbopump is provided by the introduction of roughened stator damper seals into the design.

Rotordynamics — The primary goals in the design of the CGGOT have been to provide: (1) greater than 20 percent margin over design speed for the fundamental first bending mode for low synchronous response, (2) a sufficient stability margin, and (3) a high integrity rotor balance. Meeting these provisions has required optimization of the mechanical design of the rotor, bearings, rotor support, damper seals, and housings for successful rotordynamic characteristics. The initial P&W CGGOT design moved the first fundamental rotor bending mode, with high strain energy, to well above the design speed, effectively eliminating the synchronous response due to rotor imbalance. The pitch and bounce modes of the rotor occur at 99 and 189 percent of operating speed. These modes are classified as rigid body modes and are of relatively low rotor strain energy content. They are shown in Figures 4.1.3.3-2 and -3.

Rotor bearing stiffness plays an important role in the dynamic behavior of all turbomachinery. In high-pressure rocket turbopump designs, P&W realizes the need for the combined rotor support system (i.e., bearing, carrier, and backup structure) to approach or exceed the relative stiffness of the rotor structure.

The rotor critical speed analysis has been used to set initial design requirements for each bearing stiffness. The pump end bearing is a large diameter, high load capacity ball bearing with minimal internal radial clearance (IRC) and deadband. The turbine end bearing is a large diameter, high load capacity roller bearing with a negative IRC. High rotor stiffness, coupled with stiff rotor design, without exceeding successful P&W bearing DN experience, ensure that successful rotordynamics criteria are met for this application.

Rotordynamic stability analysis will be used as a design tool to determine the final damper seal configuration requirements for optimized system dynamics. However, the CGGOT is designed such that damper seals are not critical to the dynamics of the rotor system. Each of the seals is designed for high damping, moderate stiffness, and minimal leakage. The incorporation of damper seals into the turbopump design provides: (1) reduced synchronous response throughout the operating speed range resulting in lower dynamic bearing loads and rotor deflections, (2) increased margin on the onset speed of instability (OSI), and (3) additional rotor load support between the bearings. Locations for the damper seals are being investigated to be incorporated into the next phase of design.

The CGGOT design provides for improved rotor balancing. Each major rotating component will be double piloted and indexed to the through tiebolt for positive concentricity control and balance repeatability. In addition, each major rotating component will be dynamic check balanced in detail to provide minimal residual force and moment imbalance. The dynamic balance of the rotor assembly will be completed with corrections in two planes.

The CGGOT design proposed by P&W results in acceptable rotordynamic characteristics throughout the operating range. The lightweight stiff rotor and bearing design have reduced the synchronous response due to the first fundamental rotor bending mode having been driven to more than 670 percent above the design speed. Thus, the stability of the rotor is significantly improved by avoiding the subsynchronous excitation associated with the critical speeds below 50 percent of the design speed.

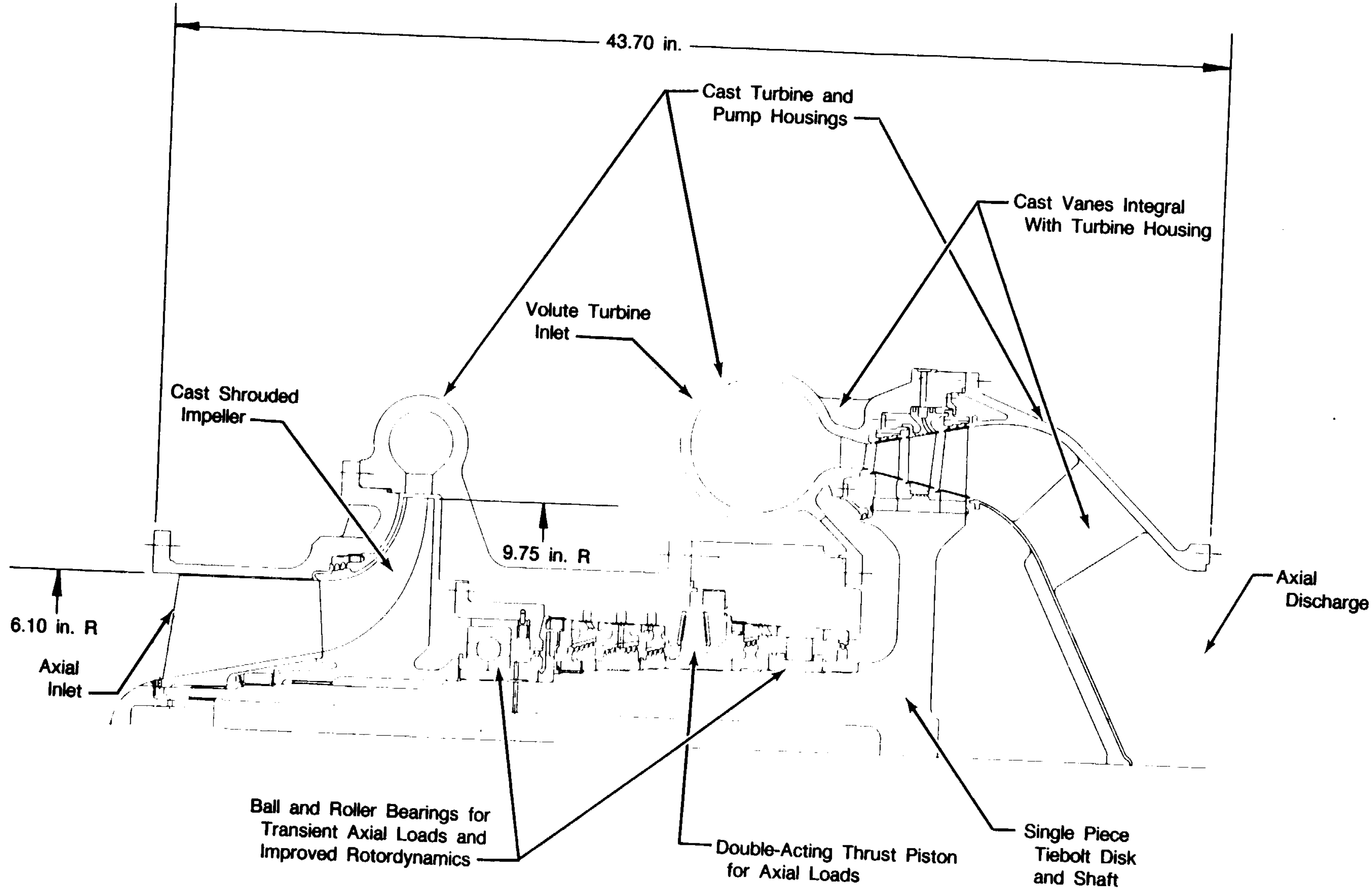

Figure 4.1.3.3-1. STBE Common Gas Generator LO_2 Turbopump

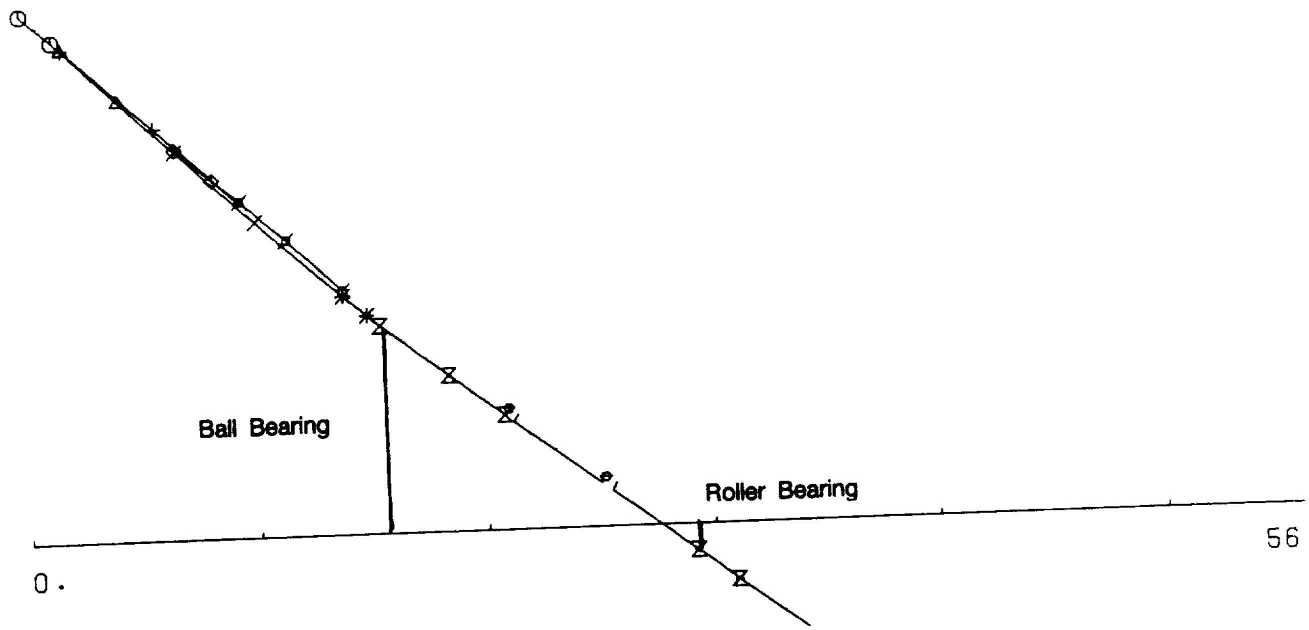

Figure 4.1.3.3-2. STBE Common Gas Generator LO_2 Turbopump Critical Speeds Analysis Showing the Pump Pitch Mode of the Rotor at 99% Design Speed (RPM = 7404)

A critical speed summary for the CGGOT is provided below.

W_{cr} (rpm)	% Design Speed	% Rotor Strain Energy	Mode Description
7404	99	4.6	Pump Pitch
14182	189	4.5	Turbine Bounce
57816	770	84.4	1st Bending

4.1.3.3.2 *Fuel Turbopump*

The mechanical description of the features of this turbopump are the same as the STBE Derivative Gas Generator Fuel turbopump. Figure 4.1.3.3-4 shows the fuel turbopump and its major components.

Rotordynamics — The three primary goals in the design of the CGGFT have been to provide: (1) a subcritical rotor design, (2) a sufficient stability margin, and (3) a high integrity rotor balance. Meeting these provisions has required optimization of the mechanical design of the rotor, bearings, rotor supports, damper seals, and housing for successful rotordynamic characteristics.

The initial phase of the CGGFT design has moved the fundamental rotor bending mode (with high rotor strain energy) to 97 percent above the design speed. Consequently, the synchronous resonant response due to the rotor imbalance is almost completely eliminated. The two modes occurring at 50 and 90 percent of the operating speed are low rotor strain energy rigid body pitch and bounce modes of the rotor, as shown in Figures 4.1.3.3-5 and -6 respectively.

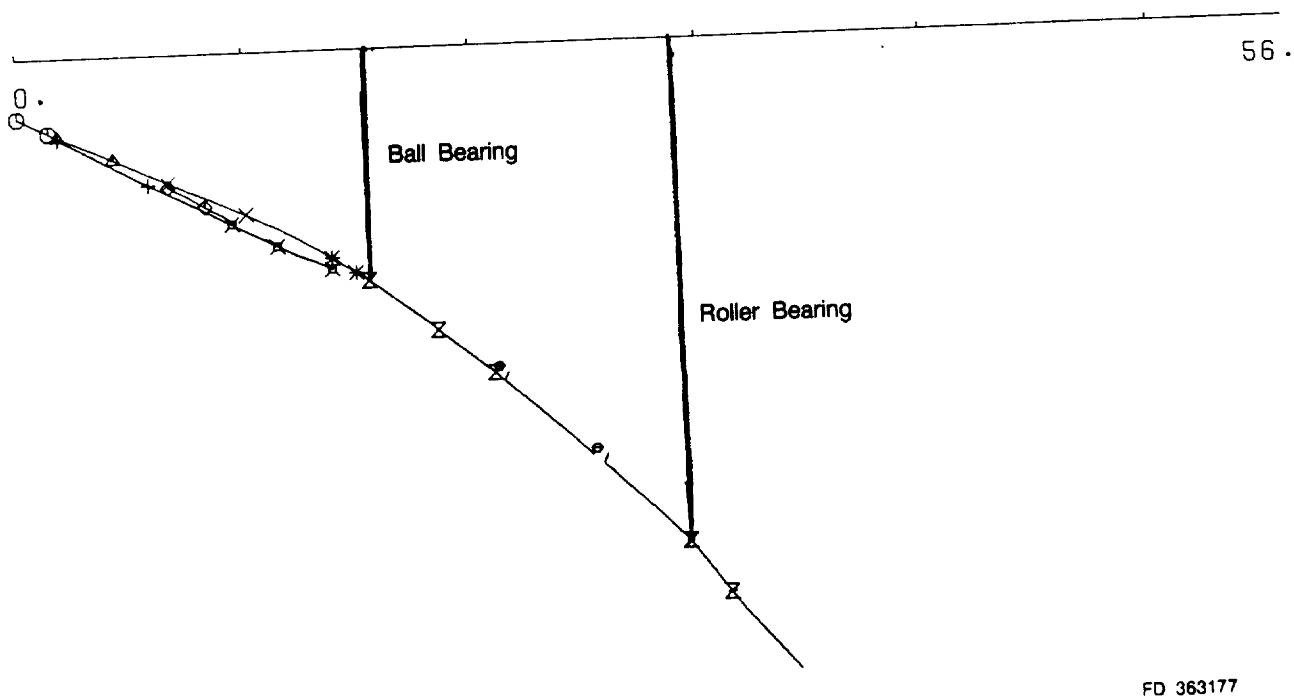

Figure 4.1.3.3-3. STBE Common Gas Generator LO_2 Turbopump Critical Speeds Analysis Showing the Turbine Bounce Mode of the Rotor at 189% Design Speed (RPM = 14182)

Rotor bearing stiffness plays a very important part in the dynamic behavior of all turbomachinery. In high-pressure rocket turbopump designs, P&W realizes the need for the combined rotor support system stiffness (bearing, carrier, and backup structure) to approach or exceed the relative stiffness of the rotor structure to minimize rotor strain energy.

The rotor critical speed analysis has been used to set initial design requirements for each bearing stiffness. The pump end bearing is a large diameter, high load capacity ball bearing with minimum IRC and deadband. The turbine end bearing is a large diameter, high load capacity roller bearing with negative IRC. Without exceeding successful P&W bearing DN experience, high rotor support stiffness coupled with a stiff rotor design in this application, ensure that successful rotordynamic design criteria are met.

Some improvements will be required in the CGGFT design of Phase A, namely, moving the rigid body rotor modes out of the operating speed range. These changes will improve the rotordynamic stability by removing the subsynchronous rotor excitation associated with speeds below 50 percent of the design speed. Also, the use of roughened stator damper seals at the impeller interstage locations will improve stability.

Rotordynamic stability analysis will be used as a design tool to determine the final damper seal configuration requirements for optimized system dynamics. The CGGFT, however, has been designed such that the damper seals are not critical to the dynamics of the rotor system. Roughened stator damper seals are included in the CGGFT design. Each of the seals is designed for high damping, moderate stiffness and minimal leakage. The incorporation of damper seals in the turbopump provides reduced synchronous response throughout the operating range resulting in low dynamic bearing loads and rotor deflections, sufficient margin on the OSI, and additional rotor load support. Parametric studies on the interstage damper seal locations will be presented in design Phase B.

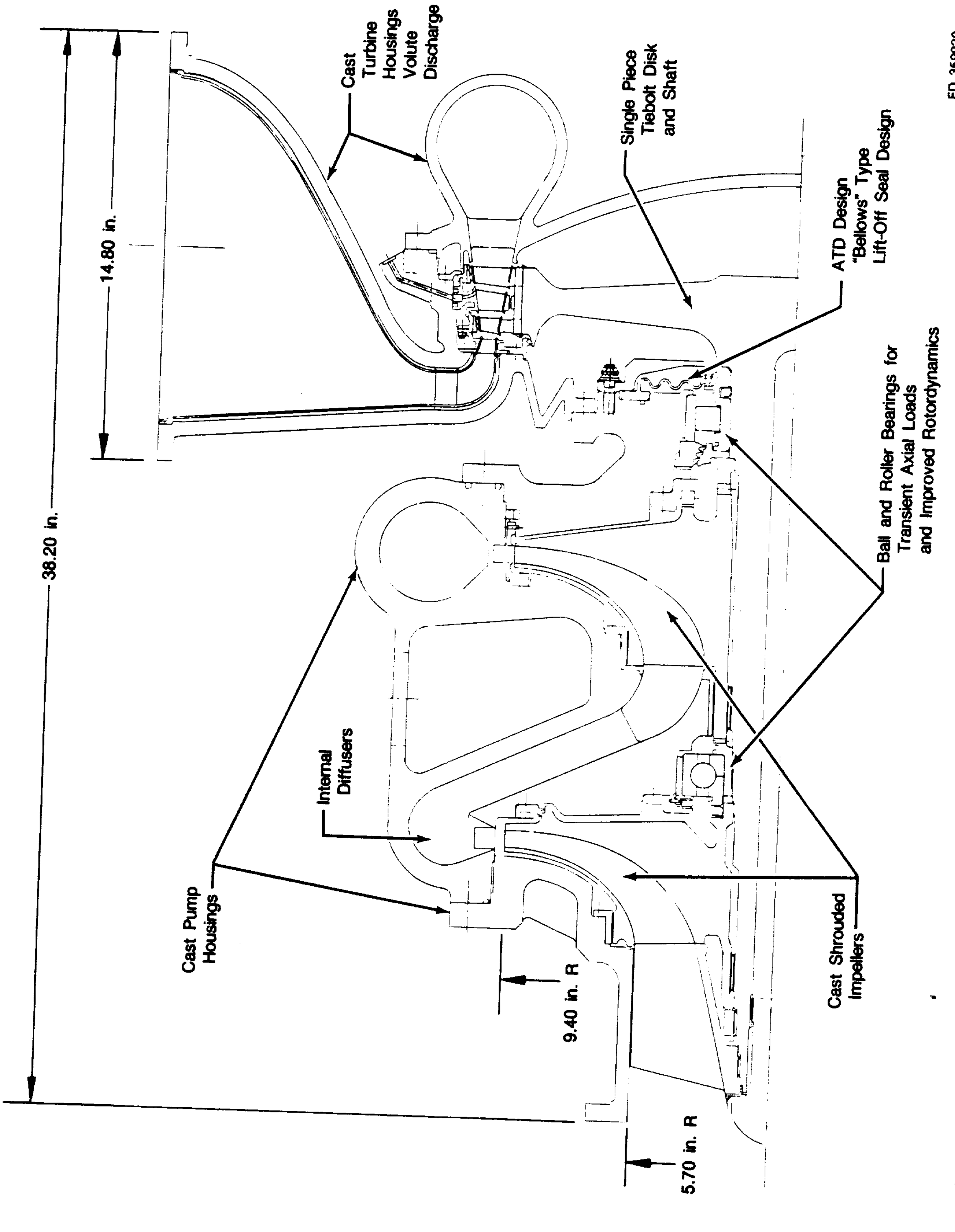

Figure 4.1.3.3-4. STBE Common Gas Generator Fuel Turbopump

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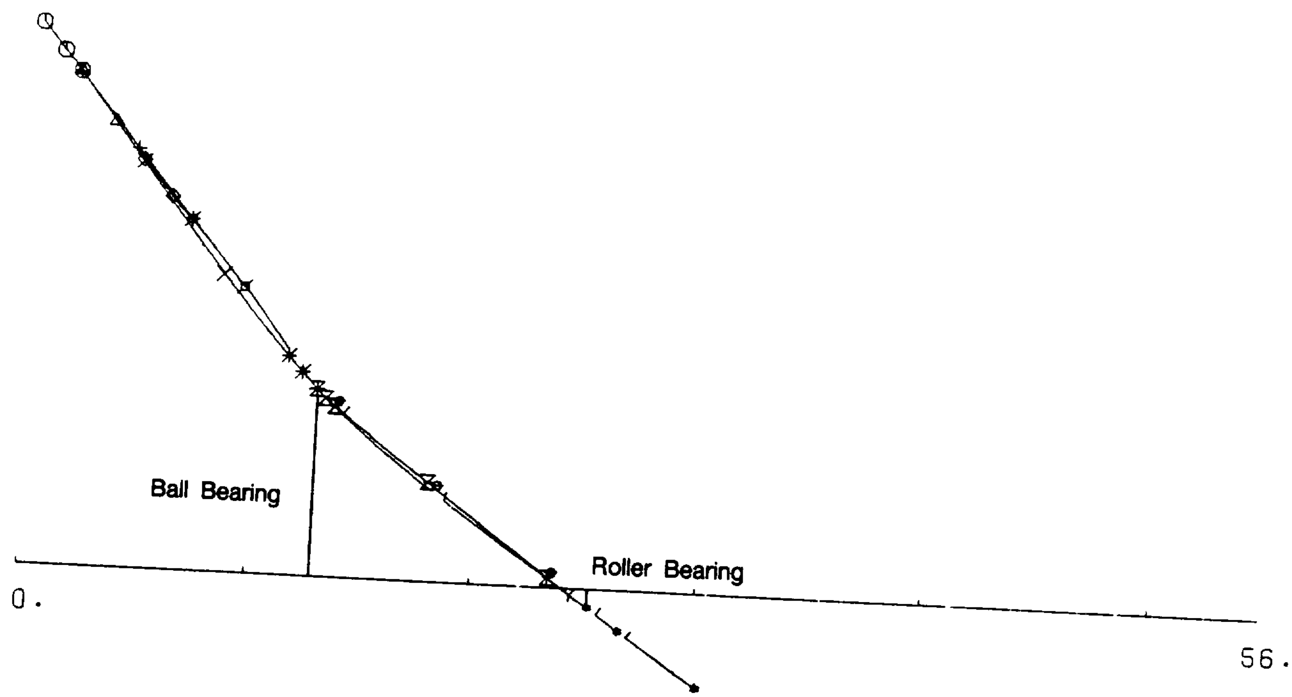

Figure 4.1.3.3-5. STBE Common Gas Generator Fuel Turbopump Critical Speeds Analysis Showing the Pump Pitch Mode of the Rotor at 50% Design Speed (RPM = 11079)

Balance provisions and techniques used for the CGGFT are identical to those used for the CGGOT.

A critical speed summary for the CGGFT is provided below.

W_{cr} (rpm)	% Design Speed	% Rotor Strain Energy	Mode Description
11079	50	10.0	Pump Pitch
18858	90	10.0	Turbine Bounce
41411	197	75.0	1st Bending

4.1.3.4 Combustor

The common core STBE/STME minimum chamber volume, injector design, and acoustic liner design were determined using the procedures outlined in section 4.1.1.4 for the derivative STBE engine. The common STBE/STME chamber and injector element designs are summarized in Table 4.1.3.4-1. The L* given in the table (31 in.) is the minimum required to meet the 98 percent characteristic velocity efficiency specified for the STBE. A slightly higher L* (37 inches) is required by the STME engine to meet a 99.3 percent efficiency. To use the STME injector in the STBE engine, higher pressure drops are required than would normally be set for injectors designed for the STBE engine alone.

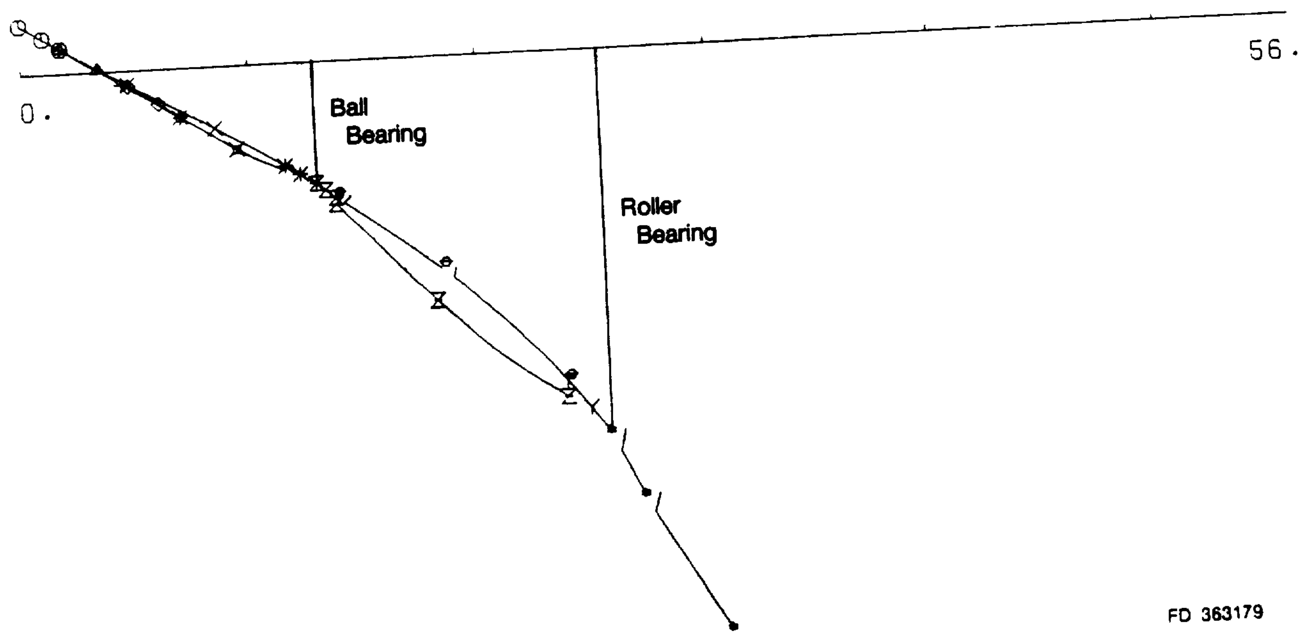

Figure 4.1.3.3-6. STBE Common Gas Generator Fuel Turbopump Critical Speeds Analysis Showing the Turbine Bounce Mode of the Rotor at 90% Design Speed (RPM = 18858)

Table 4.1.3.4-1. *Common STBE/STME Combustor and Injector Design*

	Common STBE	*STME*
Chamber L*	31.0	37.0
Fuel Flow-lb/sec	429.5	165.0
ΔP Fuel-psi	256.6	169.4
LO_2 Flow-lb/sec	1583.1	1112.1
ΔP LO_2 psi	319.3	156.4
No. of Elements	376	376
Spud ID-in.	0.448	0.448
Annular Gap-in.	0.03	0.03

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The common STBE acoustic liner design is given in Table 4.1.3.4-2. This liner will provide 27 percent acoustic absorption at the first tangential frequency (1421 Hz) of the combustion chamber.

4.1.3.4.1 Common STBE Gas Generator Main Injector

The mechanical description of the features of this main injector are the same as the STBE Derivative Gas Generator main injector, with the exception that this injector has a toroidal fuel manifold. Figures 4.1.3.4-1 through -3 show the main injector assembly, injector element and injector pattern, respectively.

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Table 4.1.3.4-2. Common STBE Acoustic Liner Design

Chamber Pressure-psi	2250
Aperture — Gas Temperature-°R	2000
Aperture — Gas Molecular Wt.	22.3
Hole Diameter-in.	0.1
Hole Length--in.	0.35
Area Ratio	0.05
Backing Cavity Depth-in.	0.60
Liner Length-in.	3.9

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4.1.3.4.2 Common STBE Gas Generator Combustion Chamber

The combustion chamber, shown in Figure 4.1.3.4-4, is regeneratively cooled by fuel from the pump discharge (high pressure). The fuel enters the thermal skin cooling jacket through the inlet manifold below the throat. The coolant then flows forward, counter to the gas path flow, to the throat. The fuel cools the chamber wall, exits at the injector interface internal manifold, and enters the injector. This flow configuration provides the coolest fuel at the throat where wall heat flux is highest.

The STBE common thrust chamber design maximizes the number of engine components that are common with the STME gas generator cycle engine. Major interface locations and diameters for the STBE were therefore maintained identical with those of the STME design. These included injector diameter, thrust chamber/regeneratively cooled nozzle interface diameter and overall thrust chamber length. Because of the basic differences in propellant performance characteristics and fuel cooling characteristics between the STBE and STME engine, it is not possible to maintain identical thrust chamber geometry and cooling passage characteristics for the two designs. The incorporation of an acoustic cavity at the forward end of the STBE thrust chamber is also a notable difference. Both designs use the entire thrust chamber fuel flow for cooling in a counterflow configuration that discharges the heated fuel directly into the injector. Table 4.1.3.4-3 summarizes the major common geometrical characteristics and differences for the two thrust chambers. The thermal design features and predicted thermal operating characteristics of the two thrust chamber designs are presented below.

4.1.3.4.2.1 STBE Common Thrust Chamber

The thrust chamber features a machined passage thermal-skin NASA-Z liner/nickel closeout assembly surrounded by a structural jacket. The chamber has an injector diameter of 20.406 inches, a throat diameter of 14.430 inches, a contraction ratio of 2.0, and extends to an expansion area ratio of 4.98:1 where it interfaces with the regeneratively cooled nozzle. The coolant passages are sized to meet the heat transfer and cycle requirements of the STBE cycle at a sea level thrust of 635K pounds and a chamber pressure of 2250 psia while maintaining the following design criteria:

- Wall thickness — 0.030 inch
- Passage land width — 0.050 inch
- Passage depth-to-width aspect ratio — 5.0
- Wall temperature — 1500 R
- Cooling enhancement from coolant passage curvature
- Cooling Mach number $\leq$ 0.5.

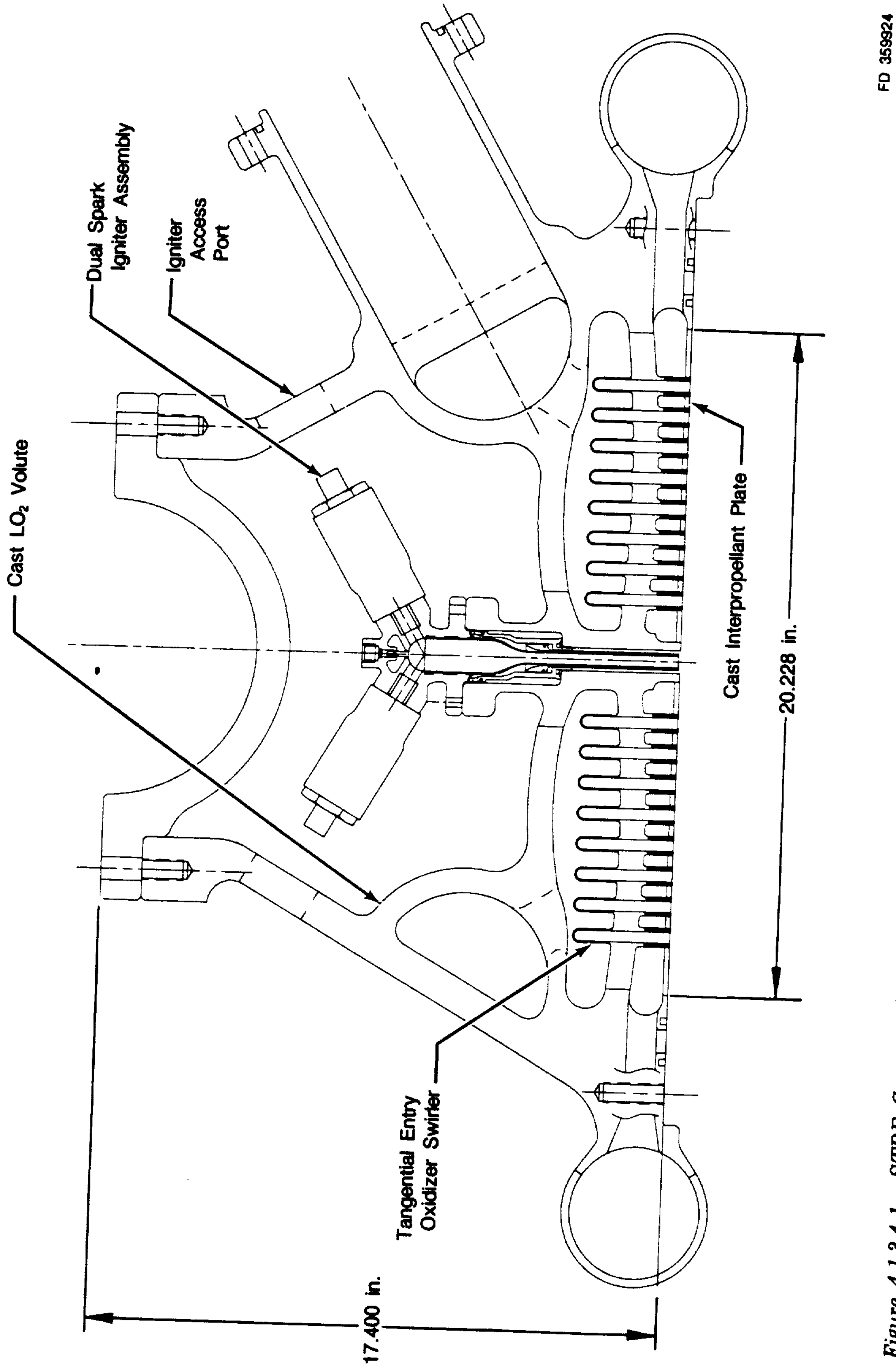

Figure 4.1.3.4-1. STBE Common Gas Generator Main Injector With Fuel Manifold

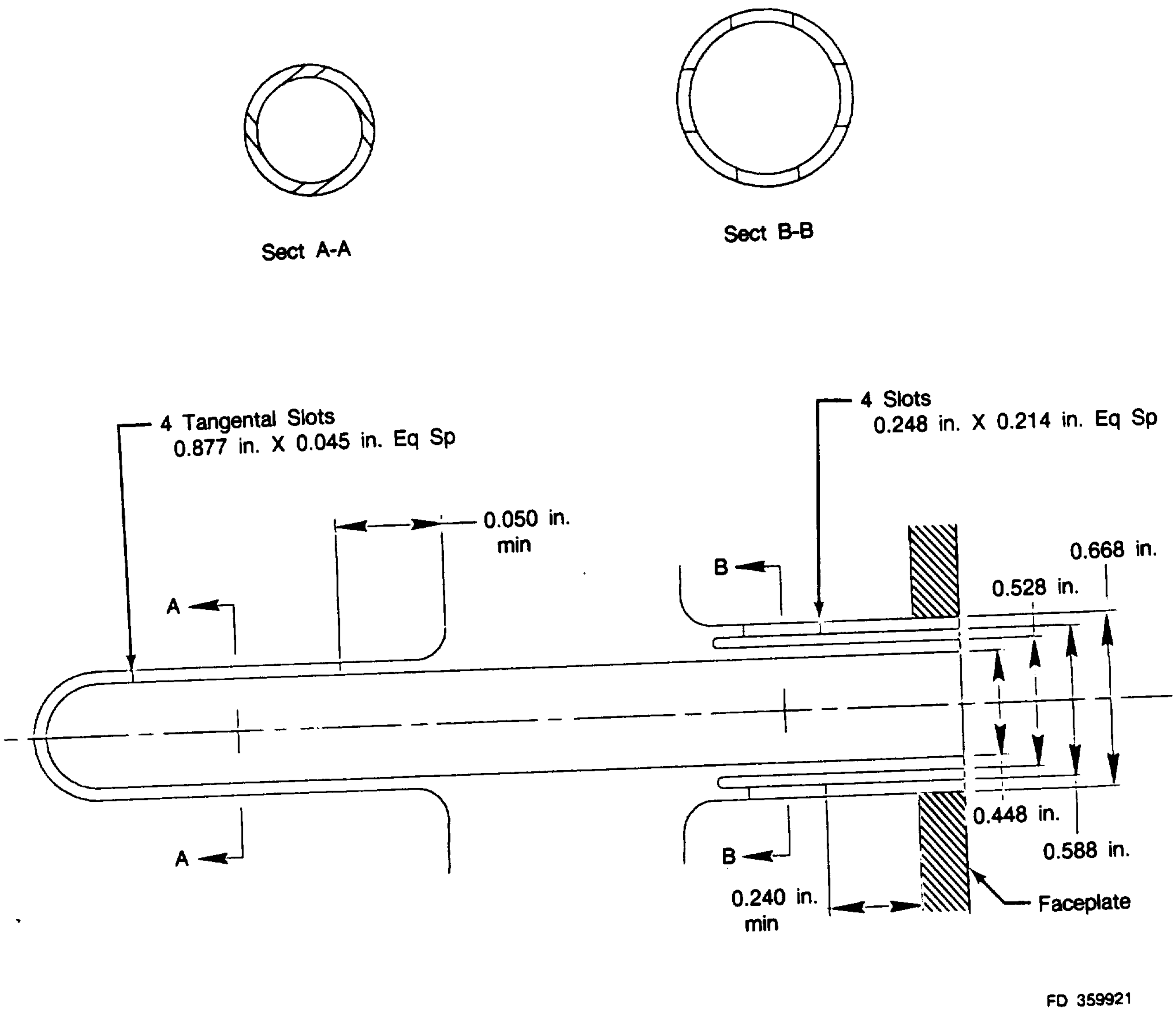

Figure 4.1.3.4-2. STBE Common Gas Generator Main Injector Element With Fuel Manifold

Figure 4.1.3.4-5 summarizes the STBE common thrust chamber contour and passage geometry.

The coolant enters the liner at the nozzle interface location at 232 R and 6600 psia and exits at the forward end of the thrust chamber at 521 R and 2631 psia after passing through the 304 chamber/acoustic cavity cooling passages. The maximum predicted heat flux to the wall, 59.1 Btu/in.^{2}-sec, occurs two inches upstream of the throat; a coolant enhancement of 7.3 percent due to curvature is predicted at this location. The maximum predicted Mach number for the coolant is 0.376. Figure 4.1.3.4-6 presents the predicted thrust chamber cooling performance at the 100 percent thrust design point.

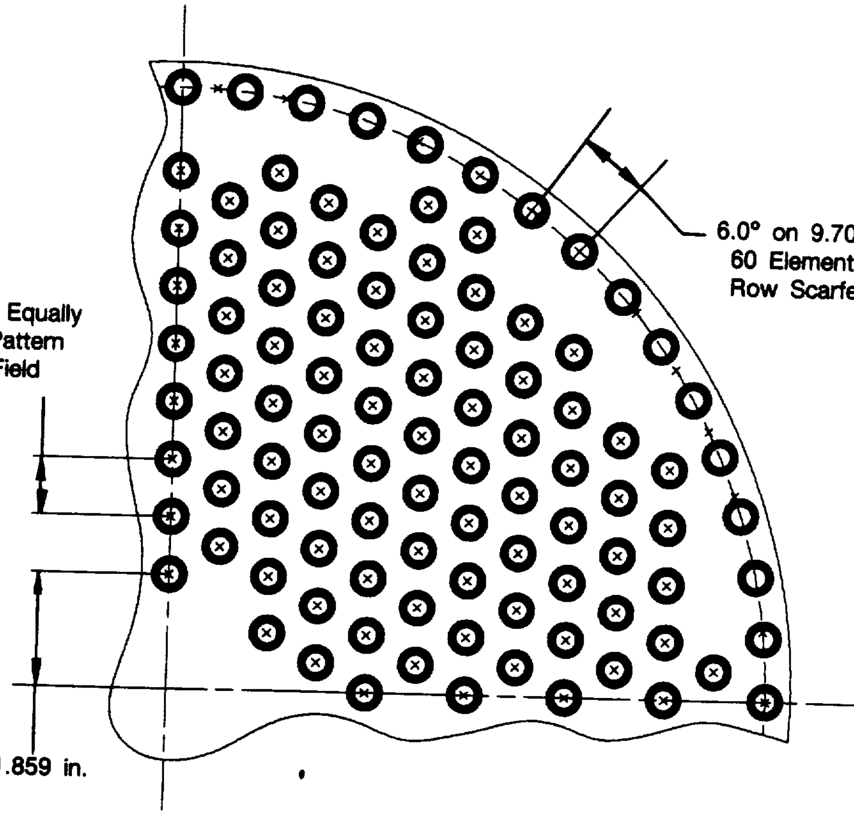

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Figure 4.1.3.4-3. STBE Common Gas Generator Main Injector Pattern With Fuel Manifold

4.1.3.4.2.2 STME Common Thrust Chamber

The STME common thrust chamber features a machined passage thermal-skin NASA-Z liner/nickel closeout assembly surrounded by a structural jacket. The chamber has an injector diameter of 20.406 inches, a throat diameter of 12.906 inches, a contraction ratio of 2.5 and extends to an expansion area ratio of 6.23 where it interfaces with the regeneratively cooled nozzle. The coolant passages are sized to meet the heat transfer and cycle requirements of the STME cycle at a vacuum thrust of 580K pounds at a chamber pressure of 2250 psia while maintaining the following design criteria:

- Wall thickness — 0.030 inches
- Passage land width — 0.050 inches
- Passage depth-to-width aspect ratio — 5.0
- Wall temperature — 1400 R
- Cooling enhancement from coolant passage curvature
- Cooling Mach number ≤ 0.5.

Figure 4.1.3.4-7 summarizes the STME common thrust chamber contour and passage geometry.

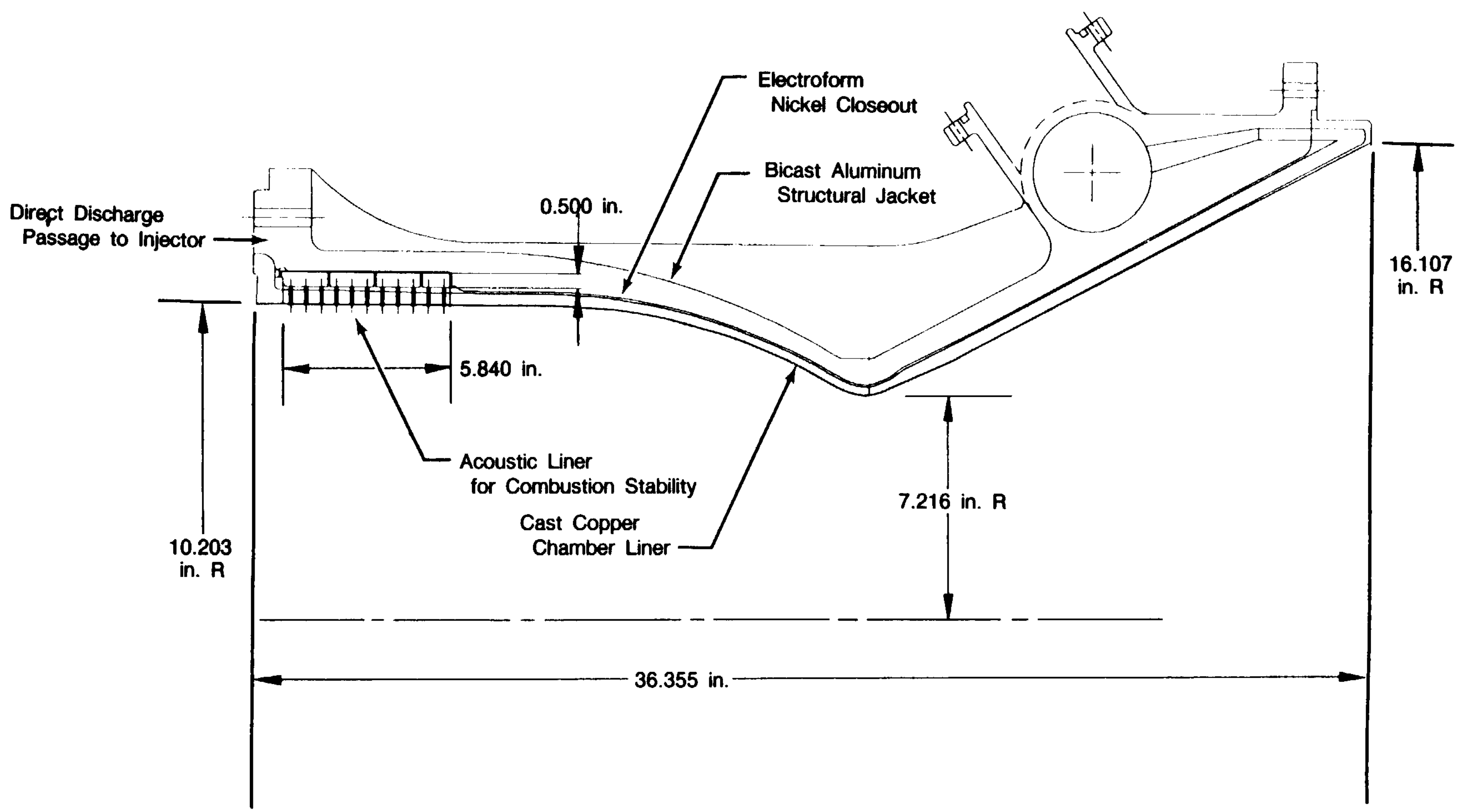

Figure 4.1.3.4-4. STBE Common Gas Generator Combustion Chamber With Inlet Manifold

Table 4.1.3.4-3. *Thrust Chamber Characteristics for STBE and STME Applications*

	STBE	*STME*
Identical Features		
Chamber Pressure — psia	2250	2250
Injector Diameter — in	20.406	20.406
Chamber/Nozzle Interface Diameter — in	32.214	32.214
Overall Length — in	36.400	36.400
Differing Features		
Chamber Length — in	22.000	20.360
Throat Diameter — in	14.430	12.906
Contraction Ratio	2.0	2.5
Characteristic Length, L* — in	39.6	43.5
Acoustic Cavity	Yes	No
Coolant	CH_4	H_2
Coolant Flow Rate — lbm/sec	334.3	133.1

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The coolant enters the liner at the nozzle interface location at 80 R and 3314 psia and exits at the forward end of the thrust chamber at 153 R and 2486 psia after passing through the 272 machined coolant passages. The maximum predicted heat flux to the wall is 62.4 Btu/in.^{2}-sec at a location one inch upstream of the throat; the enhancement of the coolant heat transfer coefficient at this location is 7.3 percent. The maximum coolant Mach number is predicted to be 0.30. Figure 4.1.3.4-8 presents the predicted thrust chamber cooling performance at the 100 percent thrust design point.

4.1.3.4.3 Torch Igniter

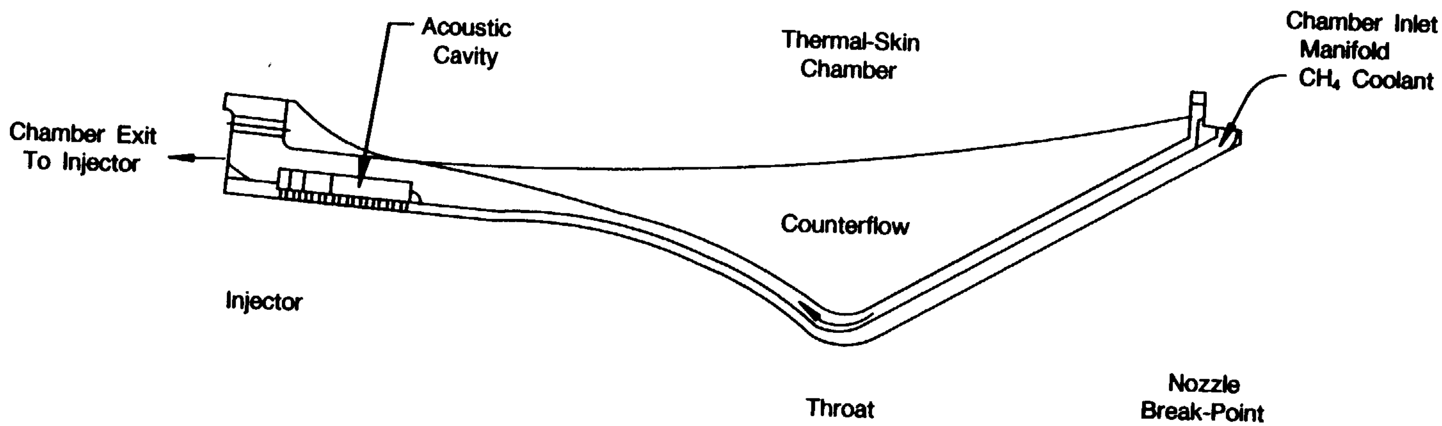
A continuous burning torch igniter was chosen for use in both the gas generator and main combustion systems because of the simplicity of the design and reliability in tests. The igniter configuration employed evolved from development efforts since 1957 at Pratt & Whitney and is based on experience gained from the successful RL10 and XRL-129 engine programs.

In the gas generator, the torch is mounted in the combustor wall, two inches axially from the injector face, and expels the hot torch combustion gases at a right angle to the flow path from the gas generator injector, thus providing safe, efficient, reliable ignition of the combustion system. In the main combustion chamber, the torch is mounted axially in the center of the injector, directing the torch down along the centerline of the combustion chamber.

The construction of the torch assembly is discussed in Space Transportation Main Engine Configuration Study, P&W FR-19830-1 Volume II, page 93.

4.1.3.4.4 Common STBE Gas Generator Combustion System

The mechanical description of the features of this gas generator combustion system are the same as the STBE Derivative Gas Generator combustion system. The gas generator assembly and injector element are shown in Figures 4.1.3.4-9 and -10, respectively.



Common Chamber Contour Data

Chamber Length = 22.00 in.
Divergent Nozzle Length = 14.36 in.
Throat Diameter = 14.43 in.
Injector Diameter = 20.406 in.
Contraction Ratio = 2.0
Divergent Nozzle Area Ratio = 4.98
L^* = 39.6 in.
ηc^* (Throat) = 0.98
Number of Passages = 304
Liner Construction - Thermal-Skin
Liner Material - NASA Z

Common Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	Passage Width (in.)	Passage Height (in.)	Land Width (in.)	Wall Thickness (in.)
-22.0	10.203	0.102	0.168	0.109	0.035
-14.9	10.203	0.102	0.168	0.109	0.035
-13.9	10.203	0.102	0.158	0.109	0.035
-11.0	10.203	0.102	0.151	0.076	0.035
-7.0	9.768	0.102	0.127	0.099	0.035
-3.0	8.449	0.102	0.104	0.075	0.035
-1.0	7.422	0.095	0.125	0.059	0.035
-0.0	7.215	0.095	0.140	0.054	0.035
3.4	9.321	0.095	0.364	0.056	0.051
7.4	11.788	0.142	0.315	0.105	0.068
11.4	14.256	0.142	0.420	0.156	0.086
14.4	16.107	0.142	0.558	0.193	0.099

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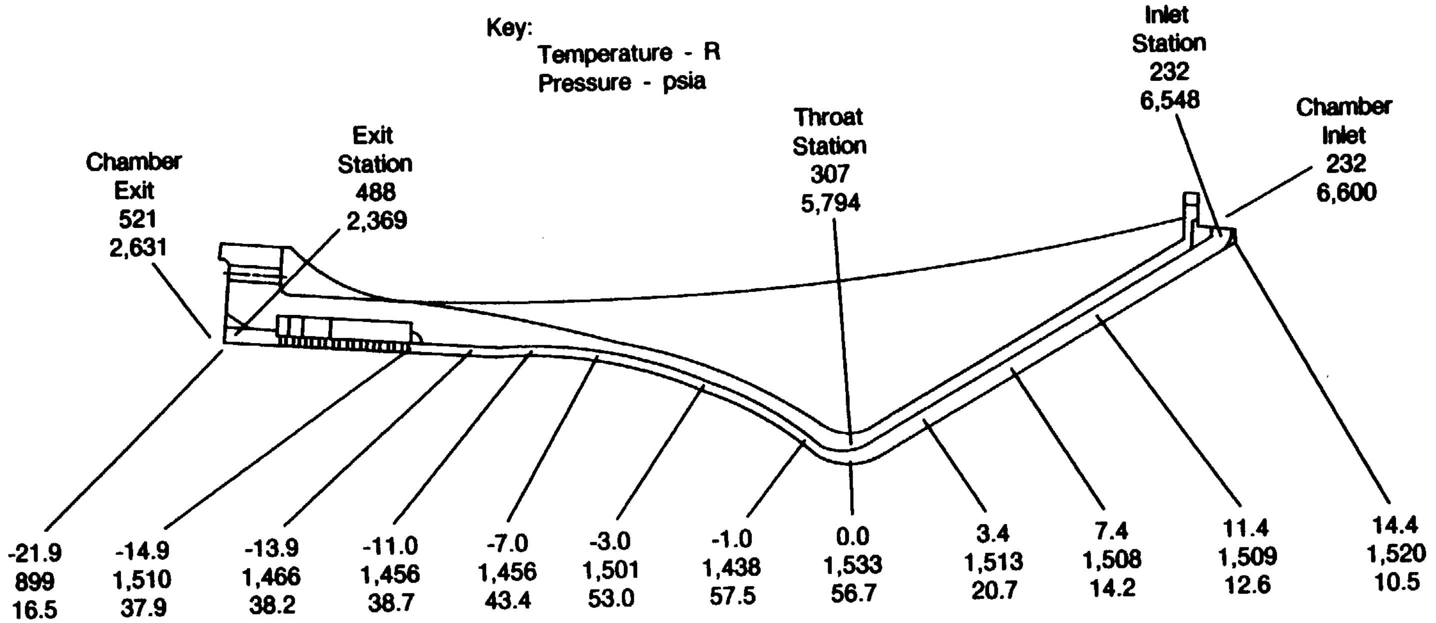
Figure 4.1.3.4-5. STBE Common Gas Generator Chamber Cooling Configuration



Coolant Performance
Thrust = 100%
M_{cool} = 334.3 lb/sec

Chamber Heat Transfer Performance

Thrust - lbf	635K
Chmber Pressure - psia	2250
Coolant Flow - lb/sec	334.3
Inlet Temperature - R	232.3
Exit Temperature - R	521.3
Coolant Heat Pickup - Btu/sec	75,943
Inlet Pressure - psia	6,600
Exit Pressure - psia	2,631
Pressure Drop - psid	3,969

Hot Wall Temperature and Heat Flux

Key:
Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2}-sec

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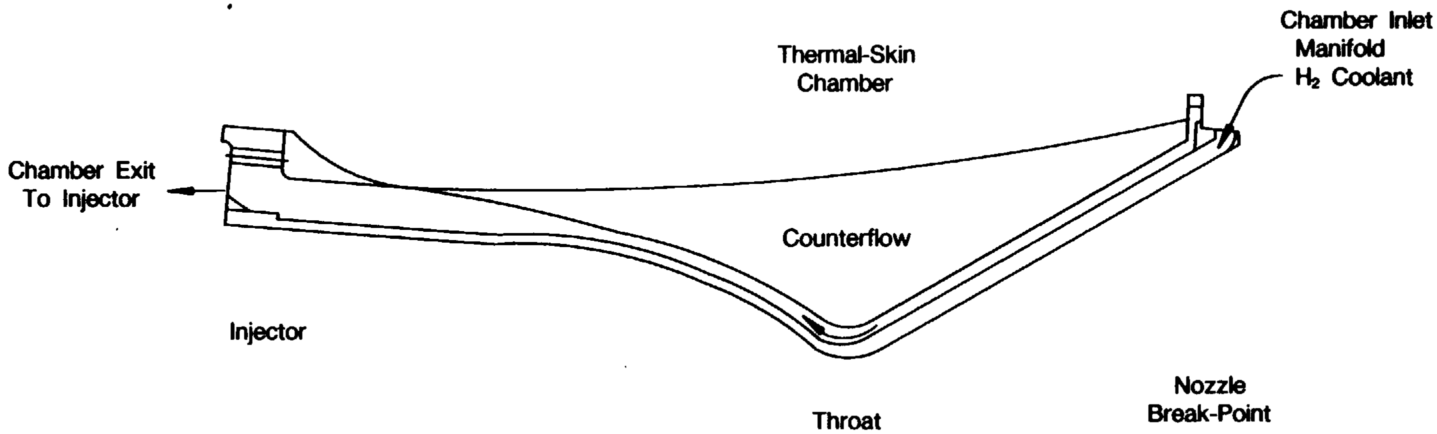
Figure 4.1.3.4-6. STBE Common Gas Generator Chamber Heat Transfer Performance Summary



Common Cooling Passage Geometry

Common Chamber Contour Data	Axial Length (in.)	Wall Radius (in.)	Passage Width (in.)	Passage Height (in.)	Land Width (in.)	Wall Thickness (in.)
Chamber Length = 20.36 in.	-20.3	10.203	0.125	0.355	0.111	0.035
Divergent Nozzle Length = 16.00 in.	-16.9	10.203	0.125	0.355	0.111	0.035
Throat Diameter = 12.906 in.	-12.9	10.203	0.125	0.318	0.111	0.035
Injector Diameter = 20.406 in.	-11.0	10.158	0.125	0.299	0.110	0.035
Contraction Ratio = 2.5	-6.0	9.154	0.125	0.246	0.086	0.035
Divergent Nozzle Area Ratio = 6.23	-4.0	8.348	0.095	0.250	0.098	0.035
L^* = 43.5 in.	-1.0	6.660	0.095	0.287	0.059	0.035
ηc^* (Throat) = 0.98	-0.0	6.453	0.095	0.260	0.086	0.035
Number of Passages = 272	4.0	8.895	0.140	0.421	0.066	0.051
Liner Construction - Thermal-Skin	8.0	11.299	0.140	0.514	0.121	0.068
Liner Material - NASA Z	12.0	13.703	0.140	0.560	0.177	0.084
	16.0	16.107	0.140	0.560	0.232	0.100

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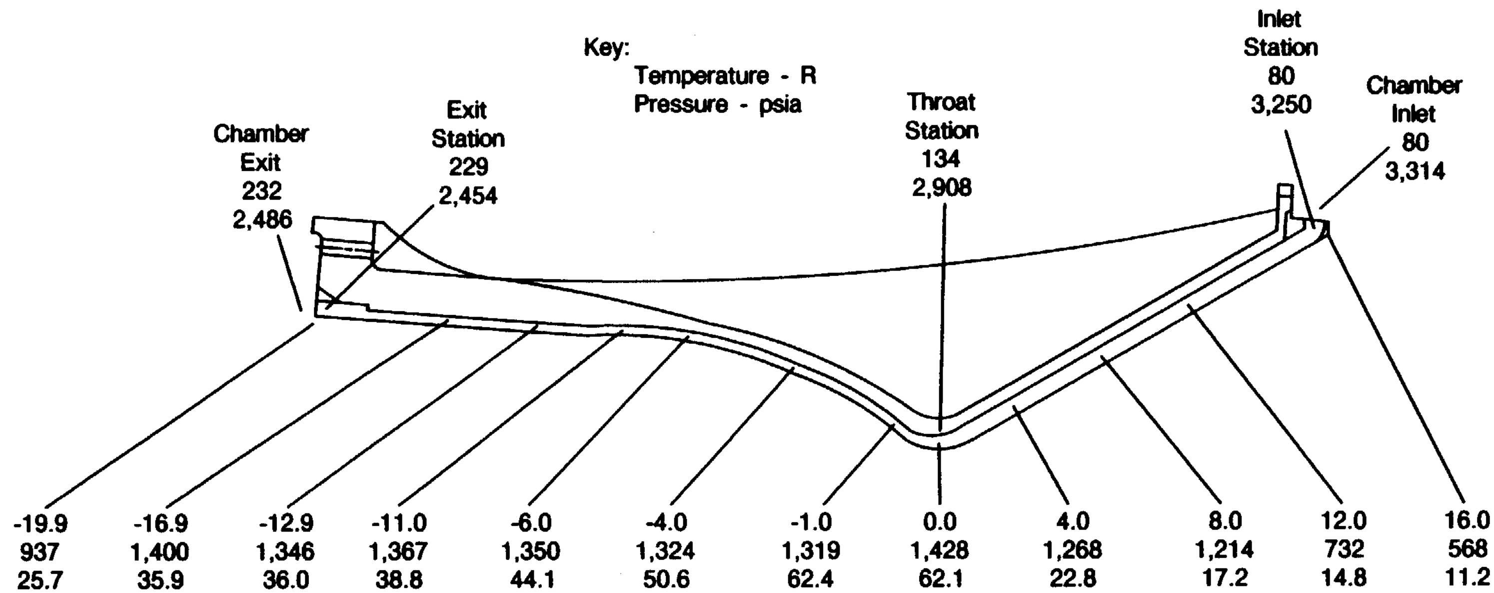
Figure 4.1.3.4-7. STME Common Gas Generator Chamber Cooling Configuration



Coolant Performance
Thrust = 100%
M_{cool} = 133.1 lb/sec

Chamber Heat Transfer Performance

Thrust - lbf	580K
Chamber Pressure - psia	2,250
Coolant Flow - lb/sec	133.1
Inlet Temperature - R	79.6
Exit Temperature - R	232.2
Coolant Heat Pickup - Btu/sec	72,938
Inlet Pressure - psia	3,314
Exit Pressure - psia	2,486
Pressure Drop - psid	828

Hot Wall Temperature and Heat Flux

Key:
Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2}-sec

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Figure 4.1.3.4-8. STME Common Gas Generator Chamber Heat Transfer Performance Summary

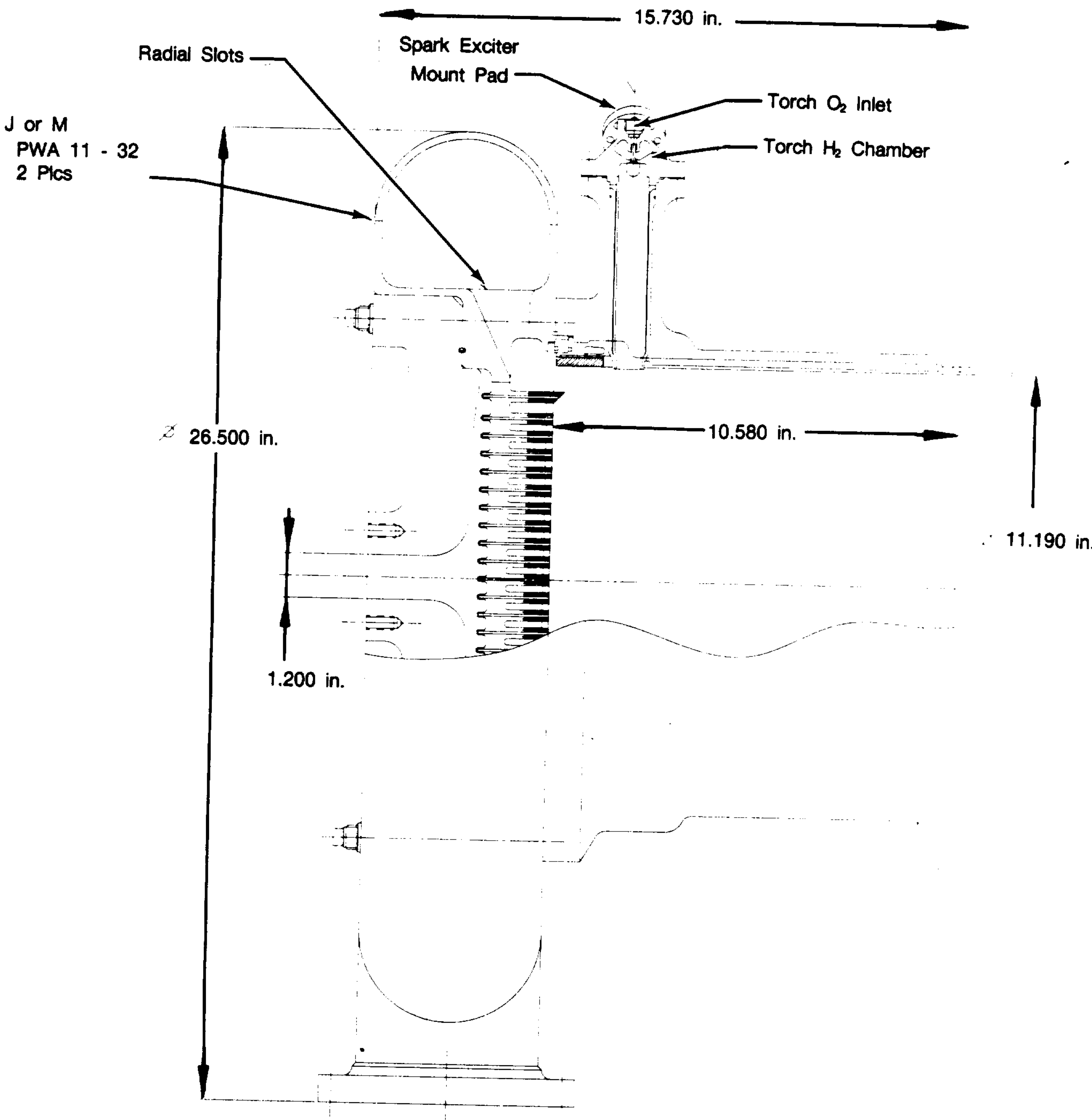


Figure 4.1.3.4-9. STME Common Gas Generator Assembly

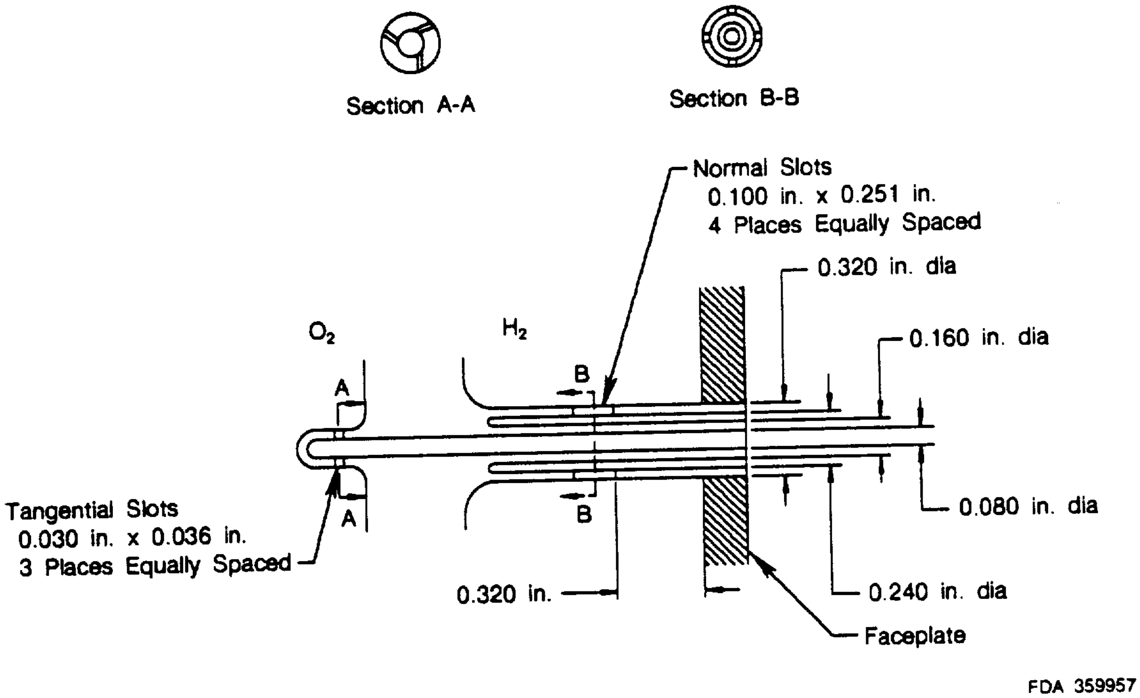


Figure 4.1.3.4-10. STME Common Gas Generator Injector Element

4.1.3.5 Nozzle

4.1.3.5.1 Common STBE Gas Generator Regeneratively Cooled Nozzle

The regeneratively cooled nozzle, shown in Figure 4.1.3.5-1, is constructed from 750 SPIF (Super Plastic Inflation Formed) tubes of AISI 347 stainless steel, surrounded by a structural shell of closed cell elastomeric foam with a filament wound composite overwrap. This shell is also designed to carry all hoop loads.

The regeneratively cooled nozzles for the STBE and STME common gas generator designs are dimensionally identical both in contour and coolant passage detail. The nozzles are constructed of 750 super plastic inflation formed (SPIF) AISI 347 stainless steel passages that simulate tubes. The nozzles are 74.1 inches long, have an inlet diameter of 32.220 inches and an exit diameter of 85.380 inches, accounting for the difference in throat diameters of the STBE and STME thrust chambers. The nozzles provide expansion from an area ratio of (4.62 or 4.98) to 35 for the STBE and from (6.23 or 6.16) to 44 for the STME. A counterflow routing of the gas generator fuel flow is used for cooling. The nozzle coolant passages were sized to meet the more demanding requirement of the STBE cycle and thermal environment at a 635K pounds sea level thrust with a chamber pressure of 2250 psia. The following design criteria were used for the nozzle:

- Wall stress — 0.2% yield stress
- Wall thickness — 0.020 in.
- Wall temperature — 1688 R (STBE)
- Ultimate tube temperature margin — 563 R (STBE)
- Coolant Mach number — 0.5.

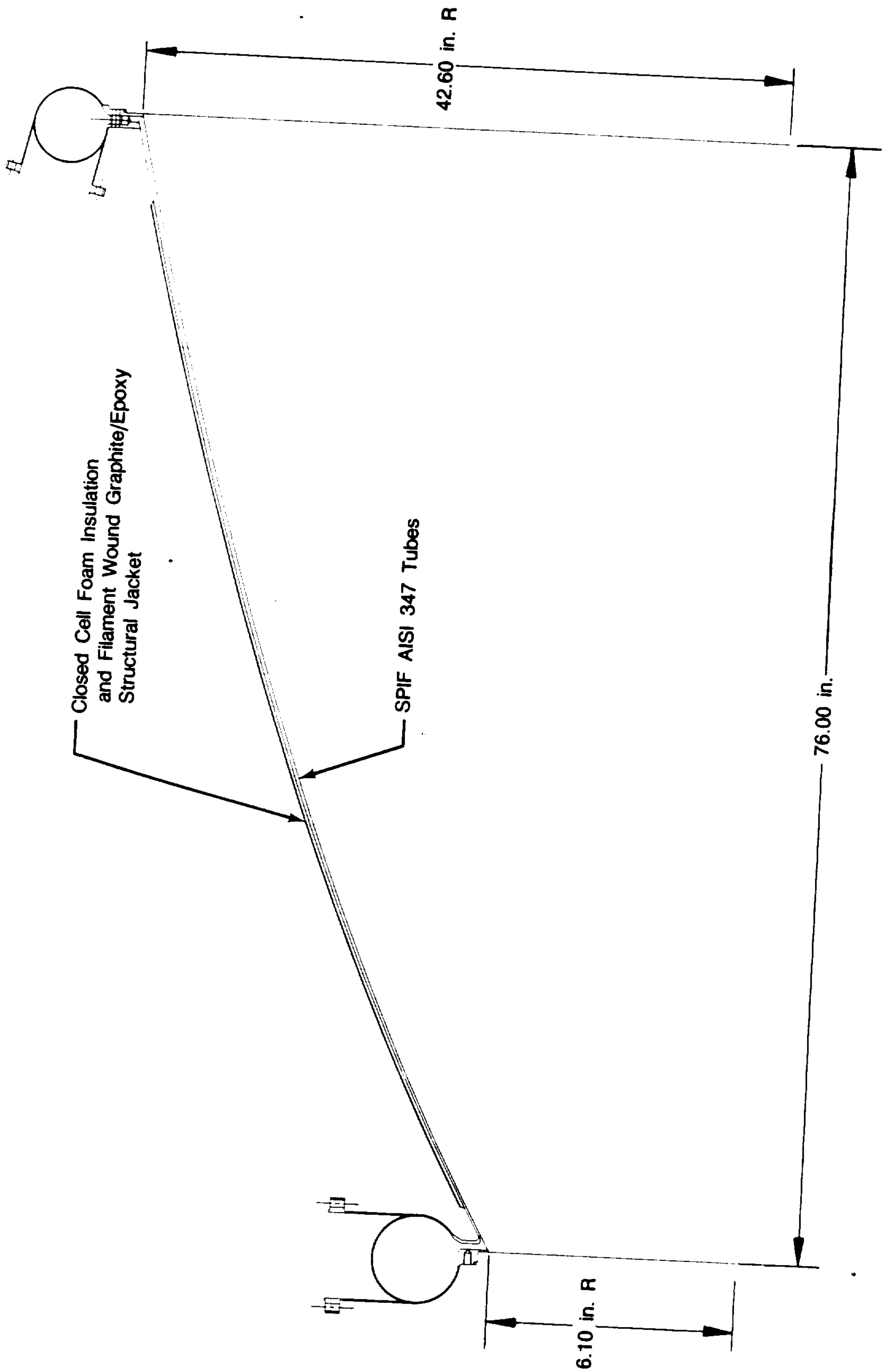


Figure 4.1.3.5-1. STME Common Gas Generator Regeneratively Cooled Nozzle

Figure 4.1.3.5-2 summarizes the regeneratively cooled nozzle geometry.

For the STBE application 105.4 lbm/sec of methane is used for cooling. The coolant enters the nozzle passages at 234 R and 4264 psia and exits at 523 R and 3338 psia. The maximum predicted hot wall temperature is 1688 R and the maximum predicted heat flux is 10 Btu/in.^{2}-sec. Figure 4.1.3.5-3 presents the predicted thermal characteristics of the regeneratively cooled nozzle under the STBE operating conditions.

Figure 4.1.3.5-4 presents similar information for the nozzle under the STME operating conditions. For this application 55.8 lbm/sec of hydrogen fuel is used for cooling. It enters the nozzle passages at 80 R and 3287 psia and exits at 298 R and 1759 psia. The maximum predicted wall temperature is 1305 R and the maximum predicted heat flux is 8.6 Btu/in.^{2}-sec.

4.1.3.6 Controls

The description of the engine controls for the Common STBE Gas Generator Engine is the same as the controls for the Derivative STBE Gas Generator Engine.

4.1.3.7 Engine Configuration and Integration

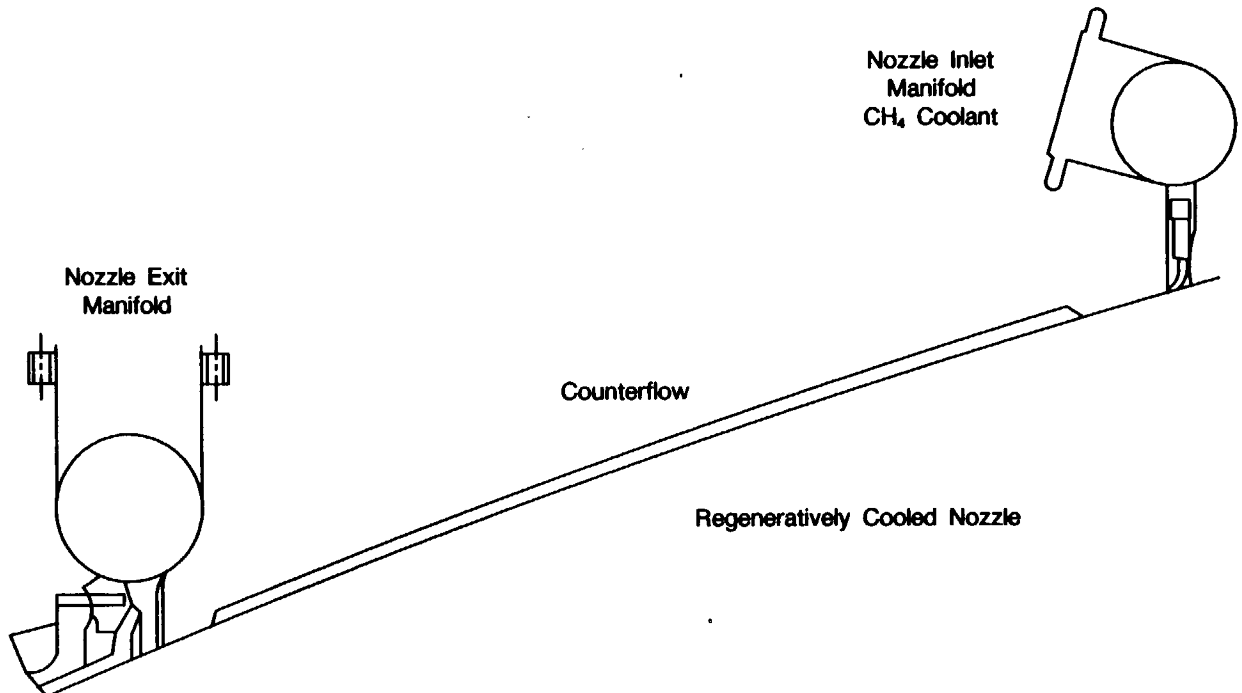
4.1.3.7.1 Common STBE Gas Generator Engine Assembly

The mechanical description of the engine assembly is the same as the STBE Derivative Gas Generator Engine Assembly. The side and top views of the engine assembly are shown in Figures 4.1.3.7-1 and -2, respectively.

4.1.3.7.2 Engine Performance

The Common STME/STBE system performance was determined during the preliminary design using the accepted JANNAF methodology. Rigorous procedures have been established for use in calculating chamber/nozzle thrust and specific impulse. The steady-state design point computer simulation provided an initial match of components and definitions of mixture ratio, mass flow, temperature and pressure levels for the detailed performance calculations using the JANNAF methodology. Figure 4.1.2.7-4 shows a flow schematic of the JANNAF performance prediction procedure followed during this task. Performance was estimated for both the main chamber flow and the gas generator flow, which is dumped overboard during engine operation. Tables 4.1.3.7-1 and 4.1.3.7-2 list the detailed performance estimates at the Design Power Level (DPL). Overall engine performance was calculated by mass weighing the main chamber flow performance with the gas generator flow performance.

During this study, detailed aerothermal analyses were made to predict component performance levels and these were incorporated into the steady-state computer model of the complete engine. Simplified flow schematics are presented in Figures 4.1.3.7-3 and -4 with key operating parameters noted. Tables 4.1.3.7-3 and 4.1.3.7-4 define performance of the individual components and their operating environments for the Common STME/STBE engines at their Design Power Levels.



Common Nozzle Contour Data

Nozzle Length - 74.1 in.
Inlet Expansion Ratio - 4.98 (6.23)**
Exit Expansion Ratio - 35.0 (43.8)**
Number of Passages - 750
Nozzle Construction - Super Plastic Inflation Formed
Nozzle Material - AISI 347
**STBE (STME)

Common Cooling Passage Geometry

Axial Length* (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)
00.0	16.11	0.121	0.128	0.020
18.6	24.85	0.199	0.205	0.020
38.8	32.71	0.265	0.269	0.020
56.4	38.11	0.311	0.312	0.020
74.1	42.69	0.349	0.349	0.020

*From Thrust Chamber Interface

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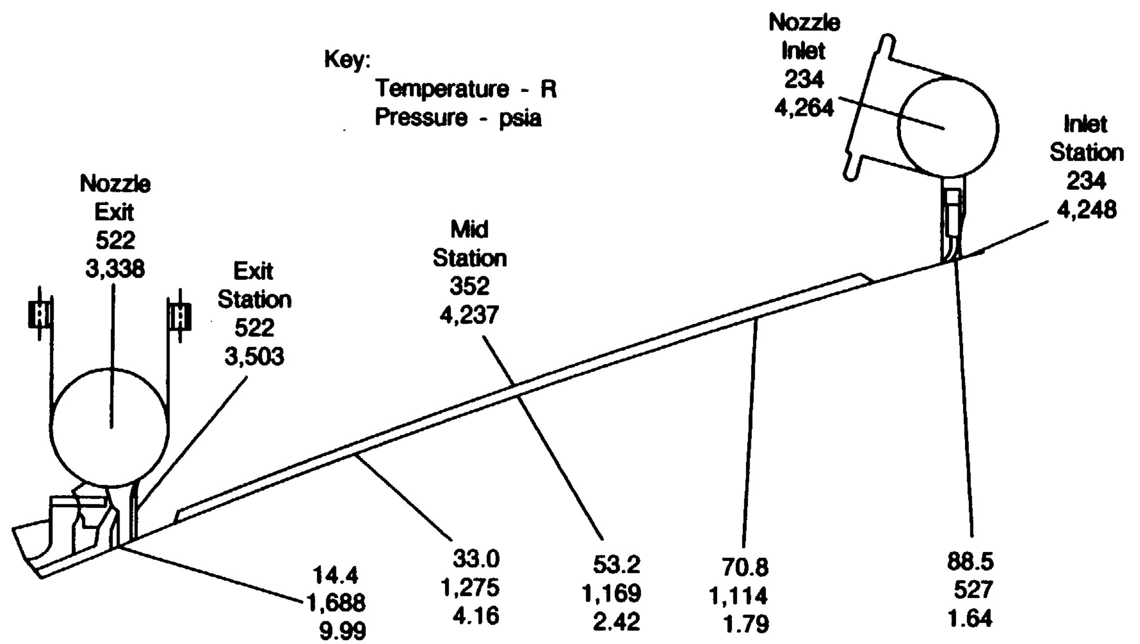
Figure 4.1.3.5-2. STBE Common Gas Generator Nozzle Cooling Configuration

Key:
Temperature - R
Pressure - psia



Coolant Performance

Thrust = 100%
M_{cool} = 205.4 lb/sec

Nozzle Heat Transfer Performance

Thrust - lbf	635K
Chamber Pressure - psia	2,250
Coolant Flow - lbm/sec	205.4
Inlet Temperature - R	233.9
Exit Temperature - R	522.5
Coolant Heat Pickup - Btu/sec	49,733
Inlet Pressure - psia	4,264
Exit Pressure - psia	3,338
Pressure Drop - psid	926

Hot Wall Temperature and Heat Flux

Key:

Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2} - sec

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Figure 4.1.3.5-3. STBE Common Gas Generator Nozzle Heat Transfer Performance During STBE Operation

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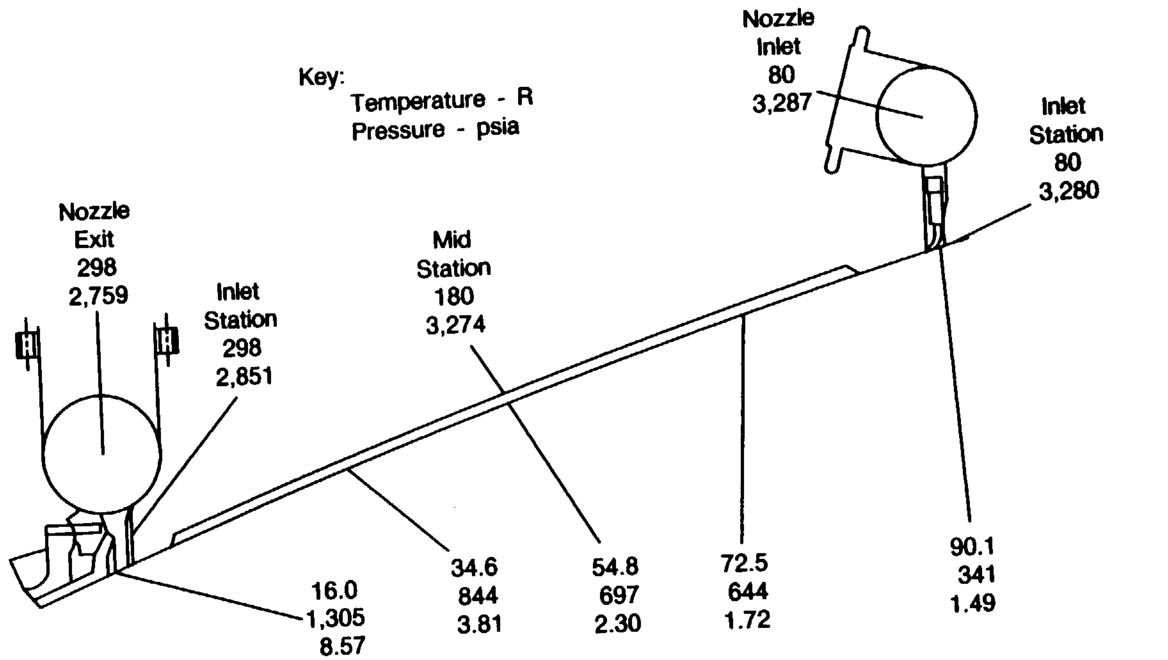

Coolant Performance

Thrust = 100%
M_{cool} = 55.8 lb/sec

Nozzle Heat Transfer Performance

Thrust - lbf	580K
Chamber Pressure - psia	2,250
Coolant Flow - lbm/sec	55.8
Inlet Temperature - R	79.8
Exit Temperature - R	297.9
Coolant Heat Pickup - Btu/sec	44,867
Inlet Pressure - psia	3,287
Exit Pressure - psia	2,759
Pressure Drop - psid	528

Hot Wall Temperature and Heat Flux

Key:

Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2} - sec

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Figure 4.1.3.5-4. STBE Common Gas Generator Nozzle Heat Transfer Performance During STME Operation

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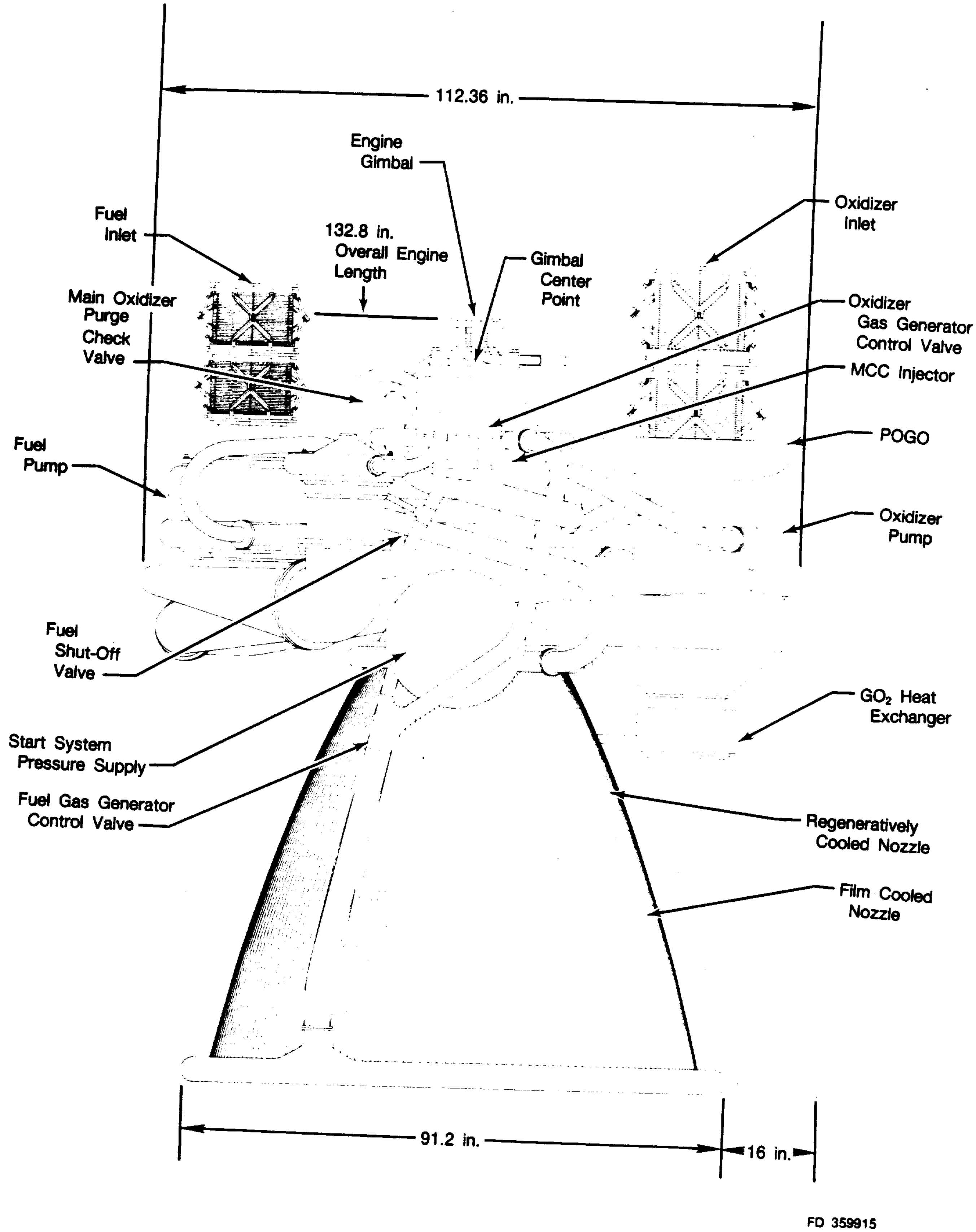

Figure 4.1.3.7-1. STBE Common Gas Generator Engine Assembly — Side View

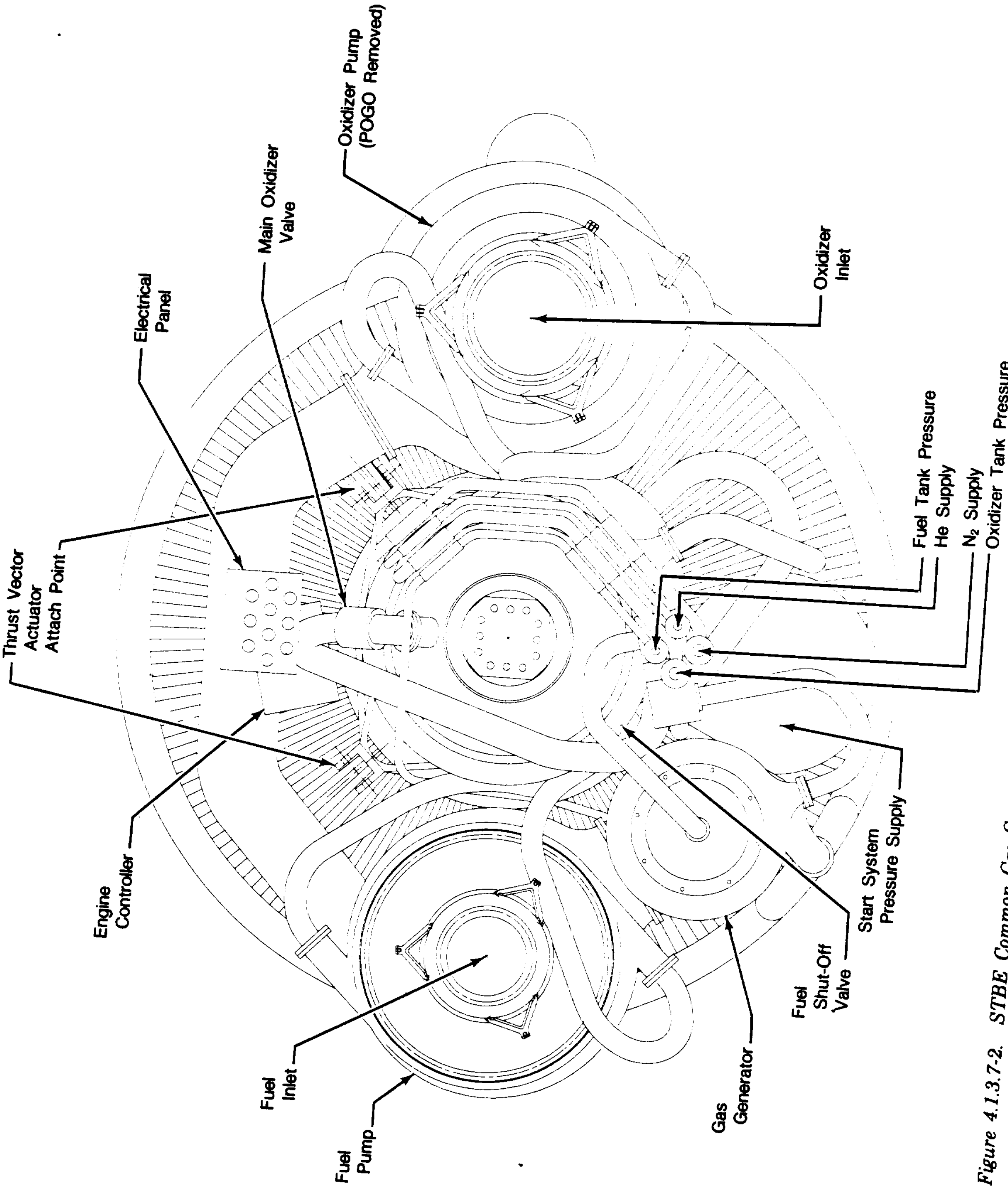

Figure 4.1.3.7-2. STBE Common Gas Generator Engine Assembly — Top View

Table 4.1.3.7-1. Common STBE Performance

	580,000-Pound Vacuum Thrust Main Chamber	Gas Generator
Pressure — psia	2250	1175
Mixture Ratio	6.65	0.838
Area Ratio	62.0	5.0
Ideal I_{sp} — sec	453.93	286.76
Δ I_{sp} ERE — sec	−3.199	0.000
Δ I_{sp} KIN — sec	−0.321	−0.338
Δ I_{sp} TDK — sec	−3.186	−4.794
Δ I_{sp} BLM — sec	−2.362	−0.598
Del. I_{sp} Vac — sec	444.86	281.03
Flowrate — lbm/sec	1279.5	38.5
Vacuum Thrust — lbf	569,173	10,826

	Overall Engine
Vacuum Thrust — lbf	580,000
Vacuum Del. I_{sp} — sec	440.07
S.L. Thrust — lbf	460,921
S.L. Del. I_{sp} — sec	349.72

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Table 4.1.3.7-2. Common STBE Performance

	635,000-Pound Sea Level Thrust Main Chamber	Gas Generator
Pressure — psia	2250	2400
Mixture Ratio	3.69	0.298
Area Ratio	35.0	5.0
Ideal I_{sp} — sec	363.59	176.01
Δ I_{sp} ERE — sec	−7.314	0.000
Δ I_{sp} KIN — sec	−0.821	−4.183
Δ I_{sp} TDK — sec	−8.060	−4.949
Δ I_{sp} BLM — sec	−1.487	−0.392
Del. I_{sp} Vac — sec	345.91	166.49
Flowrate — lbm/sec	2012.7	137.2
Vacuum Thrust — lbf	696,190	22,845

	Overall Engine
Vacuum Thrust — lbf	719,035
Vacuum Del. I_{sp} — sec	334.45
S.L. Thrust — lbf	635,008
S.L. Del. I_{sp} — sec	295.37

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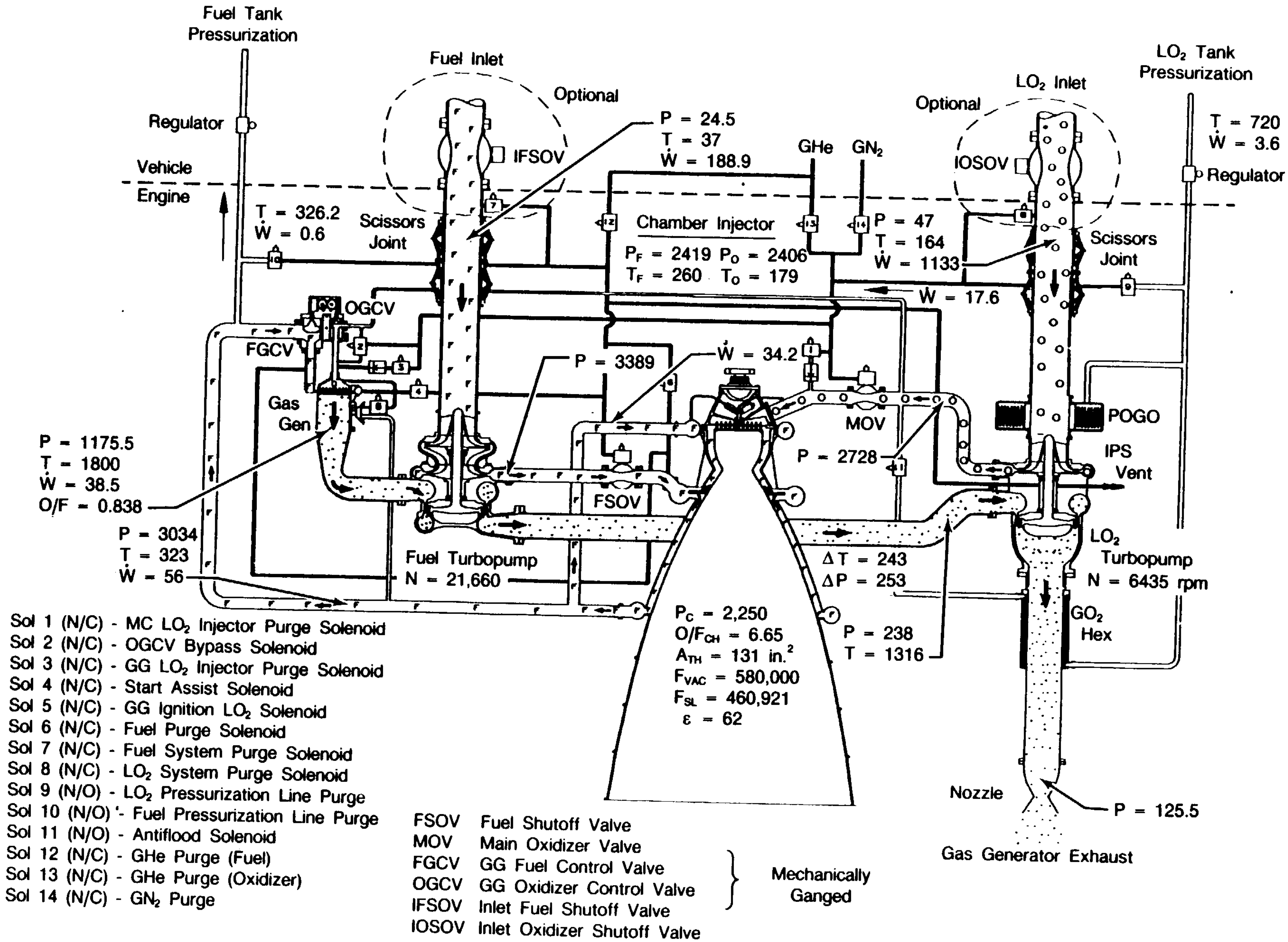

Figure 4.1.3.7-3. STBE Common Gas Generator Cycle Engine Operating Characteristics at Normal Power Level

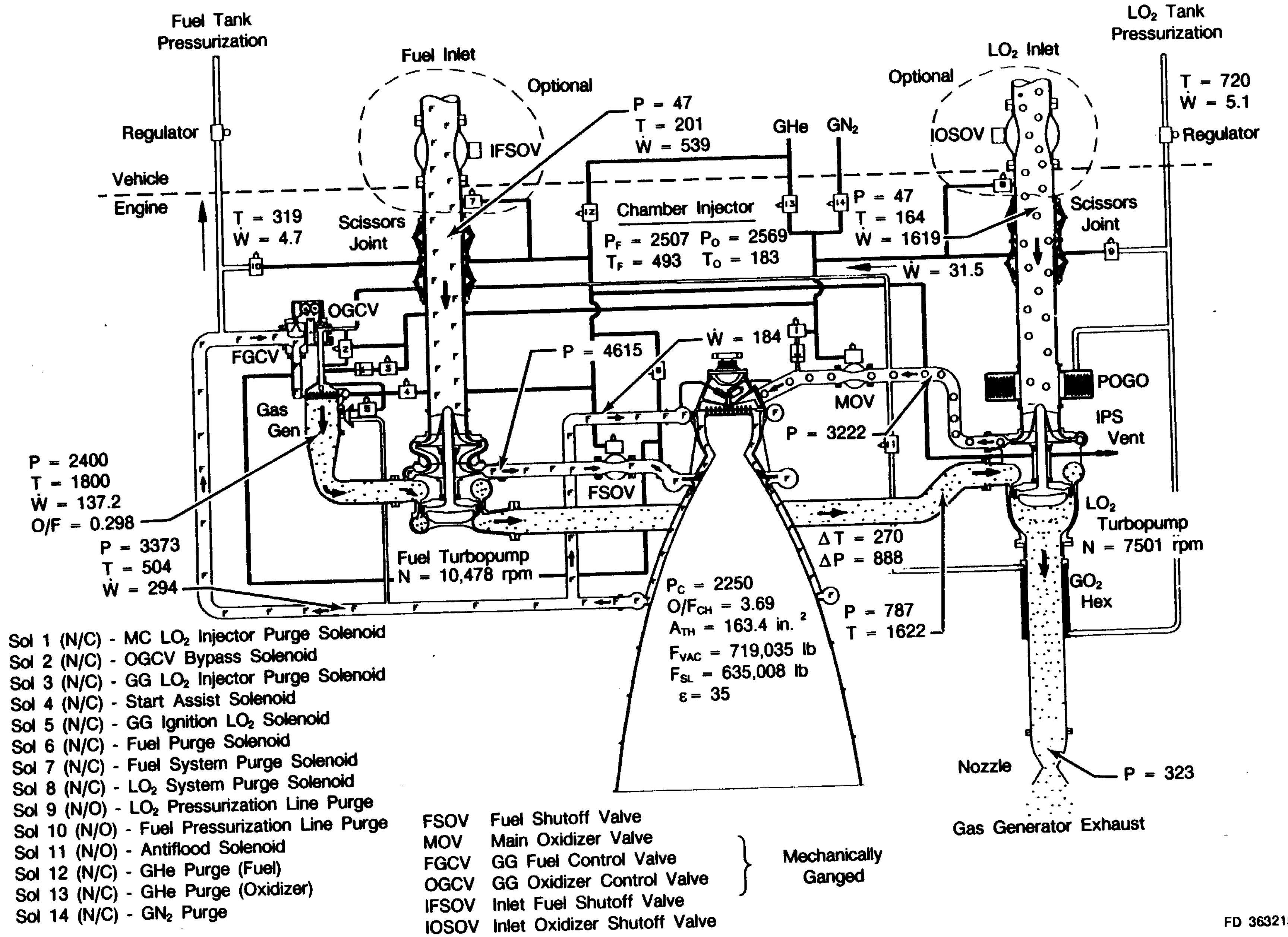

Figure 4.1.3.7-4. STBE Common Gas Generator Cycle Engine Operating Characteristics at Design Power Level

Table 4.1.3.7-3. STME Common Engine Performance — Normal Power Level

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****************************************
*           PRATT & WHITNEY            *
* GAS GENERATOR CYCLE OFF-DESIGN DECK  *
*          STME ENGINE STUDY           *
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ENGINE PERFORMANCE

VACUUM THRUST	[illegible]
SEA LEVEL THRUST	460921.
VACUUM IMPULSE	440.07
SEA LEVEL IMPULSE	349.72
TOTAL ENGINE INLET FLOW RATE	1322.1
OVERALL ENGINE MIXTURE RATIO	6.00

ENGINE HEAT TRANSFER

CHAMBER COOLANT DP	[illegible]
CHAMBER COOLANT DT	164.
CHAMBER Q	78048.
NOZZLE COOLANT DP	253.
NOZZLE COOLANT DT	243.
NOZZLE Q	51625.

CHAMBER PERFORMANCE

PRESSURE	2250.0
TEMPERATURE	6586.4
THRUST	569173.
IMPULSE	444.86
FLOW RATE	1279.5
THROAT AREA	130.74
NOZZLE AREA RATIO	62.
MIXTURE RATIO	6.65
NOZZLE EFFICIENCY	0.980
CSTAR EFFICIENCY	0.993

GAS GENERATOR PERFORMANCE

PRESSURE	1175.5
TEMPERATURE	1800.0
THRUST	10826.
IMPULSE	281.03
FLOW RATE	38.5
MIXTURE RATIO	0.838
NOZZLE EFFICIENCY	0.980
NOZZLE GAS CONSTANT	417.0
NOZZLE GAMMA	1.380
NOZZLE AREA	55.5

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	24.5	37.0	188.9	-108.2	4.39
1ST STAGE EXIT	1716.1	58.0	188.9	-10.8	4.51
MAIN PUMP EXIT	3456.5	78.3	188.9	87.8	4.61
FSOV INLET	3388.5	78.9	188.9	87.8	4.57
FSOV EXIT	3355.1	79.2	188.9	87.8	4.56
CHAM/COOL INLET	3313.5	79.6	133.1	87.8	4.54
CHAM/COOL EXIT	2448.7	243.6	133.1	674.2	1.63
CH INJ INLET	2419.4	260.2	167.3	743.6	1.52
NOZ/COOL INLET	3286.9	79.8	55.8	87.8	4.52
NOZ/COOL EXIT	3033.9	323.2	55.8	1013.7	1.49
FBP INLET	3033.9	323.2	34.2	1013.7	1.49
FBP EXIT	2448.7	324.6	34.2	1013.7	1.24
TANK PRESS OUT	2993.1	323.3	0.6	1013.7	1.48
TANK PRESS IN	24.5	326.2	0.6	1013.7	0.01
FGCV INLET	2993.1	323.3	21.0	1013.7	1.48
FGCV EXIT	1277.4	326.2	21.0	1013.7	0.69
GG INJ INLET	1264.0	326.2	21.0	1013.7	0.68

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1133.3	61.6	71.17
MAIN PUMP EXIT	2783.9	177.5	1133.3	71.4	71.47
GOX HEX IN	2728.0	177.7	3.6	71.4	71.39
TANK PRESS IN	47.0	720.0	3.6	275.4	0.22
MOV INLET	2728.0	177.7	1112.1	71.4	71.39
MOV EXIT	2455.5	178.8	1112.1	71.4	70.98
CH INJ INLET	2406.4	179.0	1112.1	71.4	70.90
OGCV INLET	2645.6	178.0	17.6	71.4	71.26
OGCV EXIT	1293.3	183.0	17.6	71.4	69.10
GG INJ INLET	1257.1	183.1	17.6	71.4	69.03

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
FUEL TURB INLET	1152.8	1800.0	38.5
FUEL TURB EXIT	250.7	1316.6	38.5
LOX TURB INLET	238.3	1316.6	38.5
LOX TURB EXIT	132.0	1169.4	38.5
NOZZLE INLET PRES	125.5		

Table 4.1.3.7-3. *STME Common Engine Performance — Normal Power Level (Continued)*

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***************************************
*            PRATT & WHITNEY          *
* GAS GENERATOR CYCLE OFF DESIGN DECK *
*           STME ENGINE STUDY         *
***************************************

       TURBOMACHINERY PERFORMANCE DATA
      *********************************
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FUEL TURBINE

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.787	0.767
HORSEPOWER	27822.	24579.
SPEED (RPM)	21660.	21660.
S SPEED	20.3	29.9
S DIAMETER	2.74	2.00
MEAN DIAMETER (IN)	21.12	21.12
VEL.RATIO (ACTUAL)	0.39	0.42
MAX TIP SPEED	2066.	2121.
BLADE HEIGHT	0.72	1.30
AN SQUARED	224.1	404.7
EFFECTIVE AREA	7.53	14.64
PRES.RATIO (T/T)	2.10	2.19
GAS CONSTANT (FT)	416.97	
GAMMA	1.3611	

FUEL PUMP

	STAGE ONE	STAGE TWO
EFFICIENCY	0.716	0.715
HORSEPOWER	26039.	26362.
SPEED (RPM)	21660.	21660.
NPSH (FT)	277.8	49358.3
SS SPEED	42221.	1121.
S SPEED	846.	827.
HEAD (FT)	54313.	54911.
DIAMETER (IN)	18.69	18.69
TIP SPEED (FT/SEC)	1768.	1768.
VOL FLOW	19329.	18778.
HEAD COEF	0.5464	0.5524
FLOW COEF	0.0700	0.0680

LOX TURBINE

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.776	0.710
HORSEPOWER	7871.	7815.
SPEED (RPM)	6435.	6435.
S SPEED	24.0	25.6
S DIAMETER	1.67	1.57
MEAN DIAMETER (IN)	26.90	26.90
VEL.RATIO (ACTUAL)	0.28	0.28
MAX TIP SPEED	801.	833.
BLADE HEIGHT	1.60	2.75
AN SQUARED	56.0	96.2
EFFECTIVE AREA	35.48	43.60
PRES.RATIO (T/T)	1.31	1.37
GAS CONSTANT (FT)	416.97	
GAMMA	1.3763	

LOX PUMP

EFFICIENCY	0.725
HORSEPOWER	15687.
SPEED (RPM)	6435.
NPSH (FT)	62.4
SS SPEED	26496.
S SPEED	850.
HEAD (FT)	5520.
DIAMETER (IN)	19.56
TIP SPEED (FT/SEC)	550.
VOL FLOW	7147.
HEAD COEF	0.5869
FLOW COEF	0.1231

VALVE DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL SHUT OFF VLV	33.3	22.88	188.9	0.98
FUEL BYPASS	585.1	1.797	34.2	19.29
FUEL GG VALVE	1715.7	0.910	21.0	57.32
MAIN OXID VALVE	272.5	11.93	1112.1	9.99
LOX GG VALVE	1352.3	0.085	17.6	51.12

INJECTOR DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL GG INJ	88.5	4.071	21.0	7.00
FUEL CH INJ	169.4	15.11	167.3	7.00
LOX GG INJ	81.6	0.350	17.6	6.49
LOX CH INJ	156.4	15.80	1112.1	6.50

***Table 4.1.3.7-4.** STME Common Engine Performance — Design Power Level*

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*******************************************
*             PRATT & WHITNEY             *
* GAS GENERATOR CYCLE OFF DESIGN DECK     *
*            STBE ENGINE STUDY            *
*******************************************
```

ENGINE PERFORMANCE	
VACUUM THRUST	710075.
SEA LEVEL THRUST	635008.
VACUUM IMPULSE	334.45
SEA LEVEL IMPULSE	295.37
TOTAL ENGINE INLET FLOW RATE	2159.6
OVERALL ENGINE MIXTURE RATIO	3.00

ENGINE HEAT TRANSFER	
CHAMBER COOLANT DP	2006.
CHAMBER COOLANT DT	265.
CHAMBER Q	58349.
NOZZLE COOLANT DP	888.
NOZZLE COOLANT DT	270.
NOZZLE Q	68215.

CHAMBER PERFORMANCE	
PRESSURE	2250.0
TEMPERATURE	6611.8
THRUST	696190.
IMPULSE	345.91
FLOW RATE	2012.7
THROAT AREA	163.43
NOZZLE AREA RATIO	35.
MIXTURE RATIO	3.69
NOZZLE EFFICIENCY	0.965
CSTAR EFFICIENCY	0.980

GAS GENERATOR PERFORMANCE	
PRESSURE	2399.5
TEMPERATURE	1800.0
THRUST	22845.
IMPULSE	166.49
FLOW RATE	137.2
MIXTURE RATIO	0.298
NOZZLE EFFICIENCY	0.970
NOZZLE GAS CONSTANT	93.6
NOZZLE GAMMA	1.174
NOZZLE AREA	43.5

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	201.0	539.9	123.1	26.40
1ST STAGE EXIT	2370.3	216.1	539.9	145.7	26.56
MAIN PUMP EXIT	4710.4	231.3	539.9	168.4	26.73
FSOV INLET	4614.6	231.9	539.9	168.4	26.68
FSOV EXIT	4567.8	232.2	539.9	168.4	26.65
CHAM/COOL INLET	4543.6	232.3	245.5	168.4	26.64
CHAM/COOL EXIT	2537.7	497.6	245.5	408.1	10.16
CH INJ INLET	2506.6	493.2	429.5	404.7	10.24
NOZ/COOL INLET	4261.9	233.9	294.4	168.4	26.48
NOZ/COOL EXIT	3373.5	503.7	294.4	400.2	12.44
FBP INLET	3373.5	503.7	184.0	400.2	12.44
FBP EXIT	2537.7	489.1	184.0	400.2	10.54
TANK PRESS OUT	3245.3	501.8	4.7	400.2	12.18
TANK PRESS IN	47.0	318.6	4.7	400.2	0.23
FGCV INLET	3245.3	501.8	105.7	400.2	12.18
FGCV EXIT	2622.6	491.0	105.7	400.2	10.75
GG INJ INLET	2600.8	490.6	105.7	400.2	10.70

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1619.7	61.6	71.17
MAIN PUMP EXIT	3335.7	180.0	1619.7	73.3	71.55
GOX HEX IN	3221.5	180.5	5.1	73.3	71.38
TANK PRESS IN	47.0	720.0	5.1	275.4	0.22
MOV INLET	3221.5	180.5	1583.1	73.3	71.38
MOV EXIT	2669.3	182.8	1583.1	73.3	70.54
CH INJ INLET	2569.3	183.2	1583.1	73.3	70.39
OGCV INLET	2956.6	181.6	31.5	73.3	70.98
OGCV EXIT	2770.4	182.4	31.5	73.3	70.70
GG INJ INLET	2656.8	182.8	31.5	73.3	70.52

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
FUEL TURB INLET	2367.2	1800.0	137.2
FUEL TURB EXIT	801.2	1623.7	137.2
LOX TURB INLET	787.4	1622.5	137.2
LOX TURB EXIT	332.7	1490.4	137.2
NOZZLE INLET PRES	323.3		

***Table 4.1.3.7-4.** STME Common Engine Performance — Design Power Level (Continued)*

PRATT & WHITNEY
GAS GENERATOR CYCLE OFF-DESIGN DECK
STBE ENGINE STUDY

TURBOMACHINERY PERFORMANCE DATA

FUEL TURBINE

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.815	0.784
HORSEPOWER	18576.	16088.
SPEED (RPM)	10478.	10478.
S SPEED	21.2	29.0
S DIAMETER	2.98	2.35
MEAN DIAMETER (IN)	21.12	21.12
VEL.RATIO (ACTUAL)	0.44	0.47
MAX TIP SPEED	999.	1026.
BLADE HEIGHT	0.72	1.30
AN SQUARED	52.5	94.7
EFFECTIVE AREA	6.60	11.15
PRES.RATIO (T/T)	1.74	1.70
GAS CONSTANT (FT)	95.20	
GAMMA	1.1587	

FUEL PUMP

	STAGE ONE	STAGE TWO
EFFICIENCY	0.715	0.715
HORSEPOWER	17307.	17357.
SPEED (RPM)	10478.	10478.
NPSH (FT)	177.7	12701.3
SS SPEED	25401.	1047.
S SPEED	843.	839.
HEAD (FT)	12615.	12644.
DIAMETER (IN)	18.69	18.69
TIP SPEED (FT/SEC)	855.	855.
VOL FLOW	9180.	9124.
HEAD COEF	0.5515	0.5528
FLOW COEF	0.1314	0.1306

LOX TURBINE

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.869	0.856
HORSEPOWER	13912.	12967.
SPEED (RPM)	7500.	7500.
S SPEED	30.6	38.6
S DIAMETER	2.30	1.95
MEAN DIAMETER (IN)	26.90	26.90
VEL.RATIO (ACTUAL)	0.46	0.48
MAX TIP SPEED	933.	971.
BLADE HEIGHT	1.60	2.75
AN SQUARED	76.1	130.7
EFFECTIVE AREA	19.04	28.54
PRES.RATIO (T/T)	1.54	1.54
GAS CONSTANT (FT)	94.54	
GAMMA	1.1667	

LOX PUMP

EFFICIENCY	0.726
HORSEPOWER	26879.
SPEED (RPM)	7500.
NPSH (FT)	62.4
SS SPEED	36921.
S SPEED	1032.
HEAD (FT)	6627.
DIAMETER (IN)	19.56
TIP SPEED (FT/SEC)	641.
VOL FLOW	10215.
HEAD COEF	0.5191
FLOW COEF	0.0825

VALVE DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL SHUT OFF VLV	46.7	22.88	539.9	1.01
FUEL BYPASS	835.8	3.343	184.0	24.77
FUEL GG VALVE	622.7	2.203	105.7	19.19
MAIN OXID VALVE	552.3	11.93	1583.1	17.14
LOX GG VALVE	186.2	0.410	31.5	6.30

INJECTOR DATA

STATION	DELP	AREA	FLOW	%DELP/P
FUEL GG INJ	201.3	4.071	105.7	7.74
FUEL CH INJ	256.6	15.11	429.5	10.24
LOX GG INJ	257.3	0.350	31.5	9.68
LOX CH INJ	319.3	15.80	1583.1	12.43

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4.1.3.7.3 Engine Costs

This section summarizes cost estimates for the 635K SL thrust, 2250 psia chamber pressure, Common STBE Gas Generator cycle. Table 4.1.3.7-5 summarizes the significant costs for the engine.

Table 4.1.3.7-5. *Common STBE Gas Generator Costs*

Total Development Cost (DDT&E), M$675*
Production Cost (TFU), M$10.7
Operations Cost/Launch/Engine, M$0.148**
Constant FY87$

*Applies to Common STBE, an additional M$1200 Development Program is estimated for the STME.
**Based on the 100th mission, 10 missions per year, and seven boosters per vehicle.

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The DDT&E Cost includes all of the functions required to design, develop, test and evaluate the engine system. All of the DDT&E functions shown in the ALS engine WBS (see Volume III) have been included. Development Cost is based on a 90-month phase C/D program with 960 engine firings for the STME and 488 for the Common STBE. Sufficient accountable firings have been included in the program to demonstrate 0.99 engine reliability with one failure.

The engine Theoretical First Unit (TFU) production cost includes all the recurring operational production cost elements specified in the ALS engine WBS. It includes manufacturing and acceptance of the Integrated Engine System, System Engineering and Integration, Program Management, Facilities Maintenance and Tooling Maintenance. The TFU estimate is based on a lot size of 100 and a 90-percent learning curve.

The Operations Cost per launch per engine includes all costs associated with the operational flight program as described in the ALS engine WBS. It includes Program Management, System Engineering and Integration, Facilities Maintenance, Operation and Support, and Training. The Operations Cost is based on a flight rate of 10 missions per year and it is the estimated cost that will be achieved after 100 total missions have been flown.

4.1.4 Unique LO_2/RP-1 Gas Generator Cycle Engine

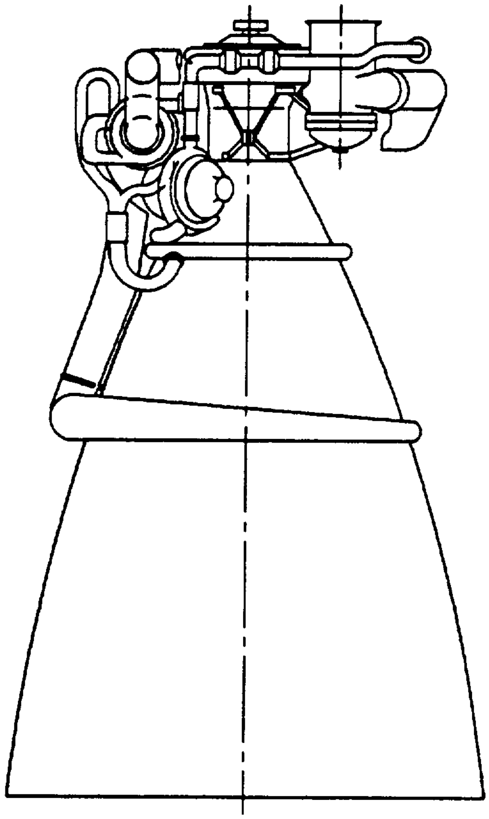
4.1.4.1 Engine Design Evolution

The LO_2/RP-1 STBE is a gas generator cycle engine with liquid oxygen and RP-1 as propellants. The engine design was initiated in the first quarter of 1988 and discussed in FR-19691-3 at 625K lb sea level thrust design.

This engine study was continued to refine the LO_2/RP-1 gas generator engine design through the last quarter of 1988. The significant changes from the initial engine design were the increase in design thrust level to 750K lb sea level and the elimination of boost pumps due to the higher vehicle supplied NPSH. The engine assembly drawing and its major characteristics are shown in Figure 4.1.4.1-1.

4.1.4.2 Engine Cycle

The candidate STBE configuration studied during Phase A is a gas generator cycle with liquid oxygen and liquid RP-1 as propellants. This engine operates at a main chamber pressure of 1501 psia at the design power level (DPL) of 750K lb sea level thrust and has the capability of running at a nominal power level (NPL) of 625,000 pounds thrust. The engine has a fixed nozzle with an area ratio of 25:1 and delivers 274.6 seconds of sea level specific impulse at DPL.



<u>Gas Generator Cycle</u>

Propellants	LO_2/RP-1
Mixture Ratio	2.75
Chamber Pressure	1500 psia
Thrust - Vacuum	863,191 lb
- Sea Level	750,000 lb
Specific Impulse - Vacuum	316.0 sec
- Sea Level	274.6 sec
Nozzle Area Ratio	25
Diameter	99 in.
Length	149 in.
Weight	TBD lb

FDA 363206

Figure 4.1.4.1-1. STBE LO_2/RP-1 Gas Generator Engine Performance Characteristics at Design Power Level

4.1.4.2.1 Flow Path Description

A simplified flow schematic for the LO_2/RP-1 STBE is presented in Figure 4.1.4.2-1, showing the major flow paths and components.

Liquid oxygen and liquid RP-1 enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps. No boost pumps are required for this system.

At the design power level, the RP-1 pump operates at 8,524 rpm to provide the RP-1 pressure levels of 2283 psia required by the cycle. From the pump exit, the RP-1 flow is split to cool the milled chamber and the tubular nozzle section separately. After cooling the nozzle, the gas generator flow is routed through a control valve and injected into the gas generator. The remainder of the nozzle coolant flow is mixed with the chamber coolant flow and is injected into the main chamber.

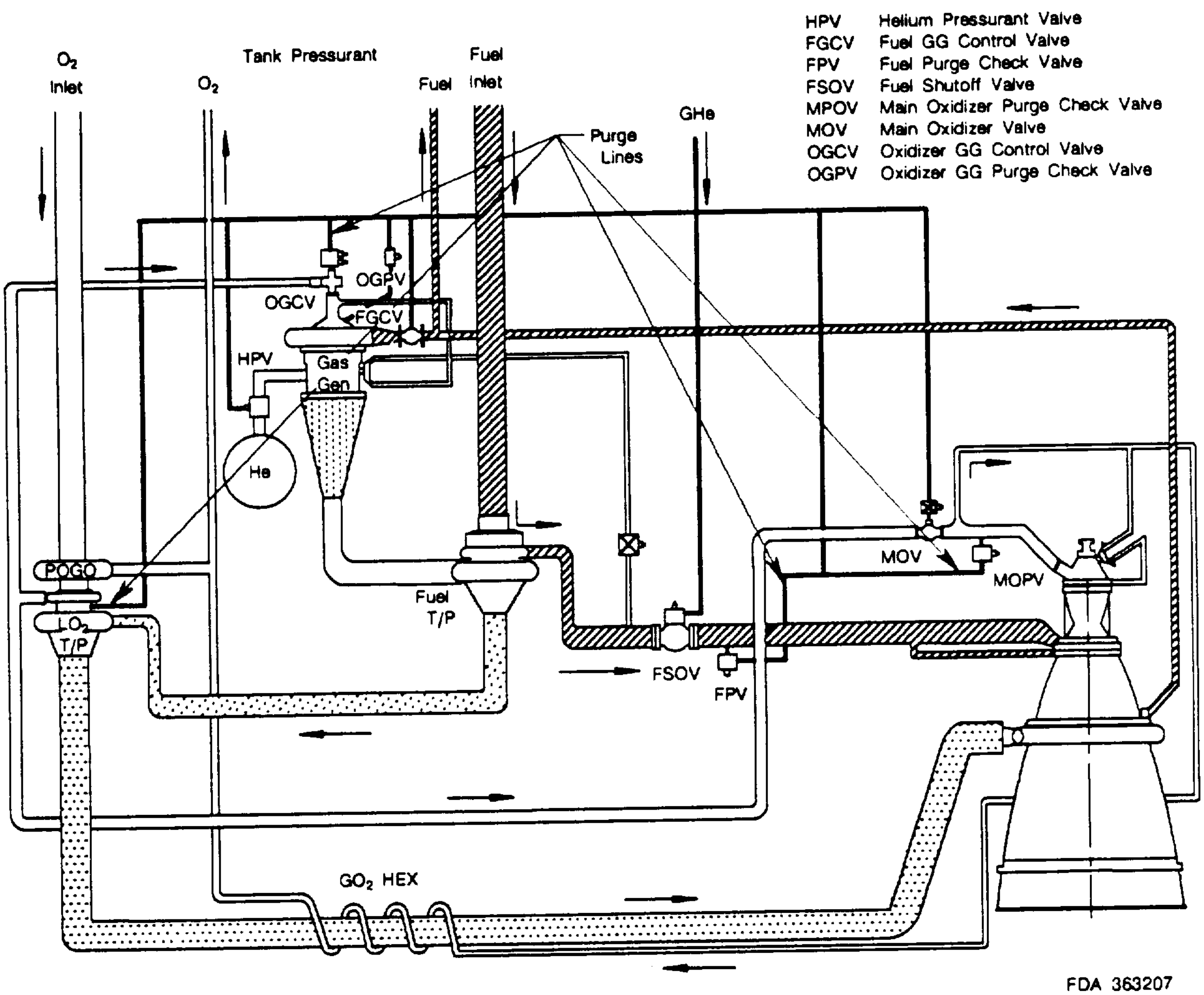


Figure 4.1.4.2-1. STBE LO_2/RP-1 Gas Generator Engine Simplified Flow Schematic

The high-pressure oxidizer pump operates at 5,645 rpm to provide the oxygen pressure level of 2091 psia required by the cycle at the design power level. From the pump exit, approximately 99 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

The high-pressure, high-temperature (1401 psia/1800 R at DPL) gas of the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas is initially expanded through the RP-1 turbine and is subsequently routed to a second turbine which powers the oxygen pump. The turbine exhaust gas is then diverted down to the nozzle below the tubular nozzle section and is used to film-cool the remainder of the nozzle from an area ratio of 20:1 to the exit area of 25:1.

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure oxidizer pump. Liquid methane enters the engine at a NPSH level, again supplied by the vehicle, sufficient for the high-speed, high-pressure methane pump; thus boost pumps are not required for this system.

At the design power level, the RP-1 pump operates at 8524 rpm to provide the methane pressure level of 2283 psia required by the cycle. From the pump exit, the RP-1 flows through the fuel shutoff valve to a split manifold, 72.0 percent of the RP-1 is used to regeneratively cool the milled channel, copper alloy main chamber from an area ratio of 3.28 back to the injector face. The remaining RP-1 flow is used to cool the tubular, stainless steel nozzle from an area ratio of 3.28 down to an area ratio of 7.85:1. This RP-1 then flows through the fuel gas generator control valve and is injected into the gas generator to combust with some of the oxygen to provide power for the high-pressure turbomachinery.

The high-pressure oxidizer pump operates at 5645 rpm to provide the oxygen pressure level of 2091 psia required by the cycle at the design power level. From the pump exit, approximately 99.2 percent of the oxygen flow is routed through the main oxidizer control valve and is injected into the main chamber. The remainder of the oxygen flows through the oxygen gas generator control valve before being injected into the gas generator.

The high-pressure, high-temperature (1400 psia/1800 R at DPL) gas of the gas generator provides the power to drive the high-pressure propellant pumps. The hot gas is initially expanded through the RP-1 turbine and is subsequently routed to a second turbine which powers the oxygen pump. From the oxidizer turbine discharge, the flow enters a heat exchanger where energy is extracted to vaporize the oxygen being provided for tank pressurization. the turbine exhaust gas is then expanded through an area ratio of 25:1 to atmospheric pressure, providing additional thrust to the overall engine output.

4.1.4.2.2 Engine Operation

The engine will be preconditioned using liquid oxygen from the tank to soak the turbopump until it is sufficiently cooled. The oxidizer inlet valve will be opened, allowing liquid from the tank to flow down to the turbopump and letting any vapors percolate back up to the tank to be vented.

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor during the oxygen phase transition from gas to liquid. The transition occurs prior to fuel injection and the fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the gas generator LO_2 injector is primed prior to opening the fuel shutoff valve to ensure liquid oxygen flow, eliminating turbine temperature spikes due to oxygen phase change. A helium spin assist is also used to initiate turbopump rotation before the fuel is introduced into the gas generator. During the start and shutdown, a small helium purge is used in the gas generator injector and main chamber injector to eliminate the danger of hot gas flow reversals during transient operation. Gas generator and main chamber ignition will be accomplished with dual electrical spark-excited torch igniters.

Main-stage engine operation is open-loop controlled. The fuel gas generator control valve (FGCV), the oxygen gas generator control valve (OGCV), and the main oxidizer valve (MOV), shown in Figure 4.1.4.2-1, are used to set the engine thrust and mixture ratio. Thrust and main chamber mixture ratio are set on the ground by trimming the MOV and OGCV respectively. The gas generator mixture ratio is set using the FGCV. All valves are operated by hydraulic actuators.

Engine acceleration is accomplished by a time-based scheduling of the valves to the commanded starting level (~ 20 percent power level). The acceleration to full thrust is also accomplished with open-loop valve schedules. Engine shutdown is accomplished using a time-

based scheduling of the propellant valves. The OGCV is closed first to power down the turbopumps, then the MOV closes, followed by shutting off the RP-1 system.

In addition to a normal operational mode, the engine system is capable of shutdown resulting from detected problems or LO_2 starvation at the end of the burn duration.

4.1.4.3 Combustor/Thrust Chamber

The STBE RP-1/LO_2 gas generator thrust chamber features a machined passage thermal-skin NASA-Z liner/nickel closeout assembly surrounded by a structural jacket. The coolant enters the inlet manifold and flows toward the injector, where it discharges directly into the injector. The chamber inlet manifold is common with the tubular nozzle which improves the inlet geometry and reduces inlet pressure drop. The chamber is cooled with 87 percent of the chamber fuel flow, which discharges directly into the injector. The remainder of the chamber fuel flow comes from the regeneratively cooled nozzle and enters the injector through a small manifold. The thrust chamber has a throat diameter of 19.81 inches, an injector diameter of 31.32 inches and a contraction ratio of 2.5. The inclusion of the acoustic liner in the chamber has increased the difficulty of cooling the liner with the chamber. To cool the liner within the cycle requirements, the number of passages has been set at 370 with a maximum passage height/width aspect ratio of 5.0. The cooling at the throat has been further improved by designing for coolant side curvature enhancement of the heat transfer film coefficient. Figure 4.1.4.3-1 summarizes the thrust chamber contour and passage geometry.

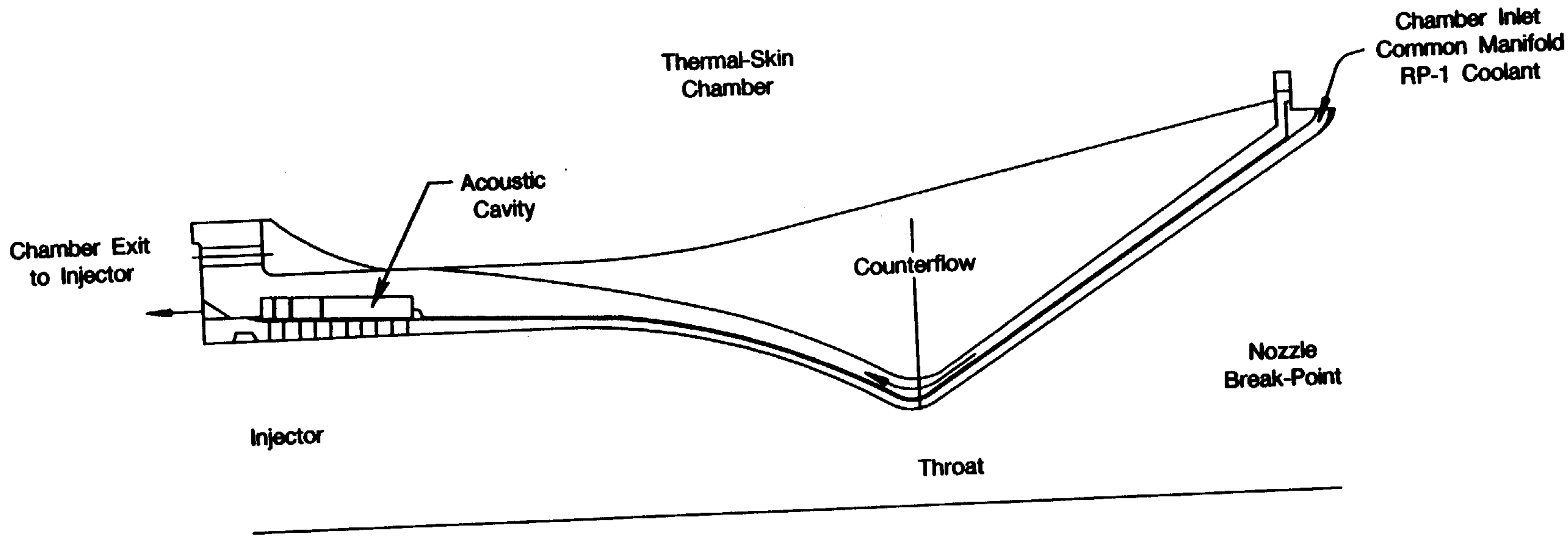
The coolant passage dimensions have been sized to meet the heat transfer, cycle and fuel coking requirements at the 120 percent thrust design point of 750K lbf at a chamber pressure of 1500 psia. The coolant split between the chamber and the nozzle has been set so that the coolant exit temperatures are approximately equal at 760 R. Figure 4.1.4.3-2 presents the predicted thrust chamber cooling performance at the 120 percent thrust design point. The chamber liner has been designed so that the maximum hot wall temperature is approximately 1525 R. The maximum wall heat flux at this wall temperature is 38.2 Btu/in.^{2}-sec which occurs one inch forward of the throat. The coolant side curvature enhancement at the high heat flux point is approximately 35 percent. The coolant enters the liner at 575 R and 2702 psia and exits at 775 R and 1874 psia.

4.1.4.4 Nozzle

4.1.4.4.1 Regeneratively Cooled Nozzle

Figure 4.1.4.4-1 summarizes the regeneratively cooled nozzle geometry. The nozzle is constructed of 630 super plastic inflation formed AISI 347 stainless steel passages that simulate tubes. The nozzle is 20 inches long and extends from an expansion area ratio of 3.28:1 to an exit area ratio of 7.85:1. The number of passages and the passage diameters have been sized so that the operating stresses of the wall never exceed the 0.2 percent yield stress. An alternate nozzle design could be constructed of 670 Haynes 230 tubes.

Figure 4.1.4.4-2 presents the predicted heat transfer performance of the nozzle. At the design point of 120 percent thrust the nozzle is cooled with 201 lbm/sec of fuel that enters at 569 R and 2303 psia. The coolant exits at 773 R and 2251 psia. The maximum hot wall temperature and heat flux is 1572 R and 9.9 Btu/in.^{2}, respectively.



Chamber Contour Data

Chamber Length = 20 in.
Divergent Nozzle Length = 15 in.
Throat Diameter = 19.81 in.
Injector Diameter = 31.32 in.
Contraction Ratio = 2.5
Divergent Nozzle Area Ratio = 3.28
L^* = 41.0 in.
η_c^* (Throat) = 0.95
Number of Passages = 370
Linear Construction - Thermal-Skin
Linear Material - NASA Z

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	Passage Width (in.)	Passage Height (in.)	Land Width (in.)	Wall Thickness (in.)
-20.0	11.89	0.157	0.460	0.110	0.040
-16.0	11.89	0.157	0.460	0.110	0.040
-14.0	11.89	0.161	0.460	0.110	0.040
-12.0	11.88	0.161	0.460	0.110	0.040
- 9.2	11.43	0.161	0.460	0.102	0.040
- 5.0	10.34	0.108	0.332	0.085	0.040
- 3.0	11.64	0.108	0.324	0.089	0.040
- 0.0	9.90	0.109	0.324	0.060	0.040
2.0	10.97	0.113	0.324	0.073	0.048
5.0	12.56	0.113	0.324	0.100	0.061
10.0	15.22	0.160	0.480	0.096	0.080
15.0	17.89	0.160	0.480	0.143	0.100

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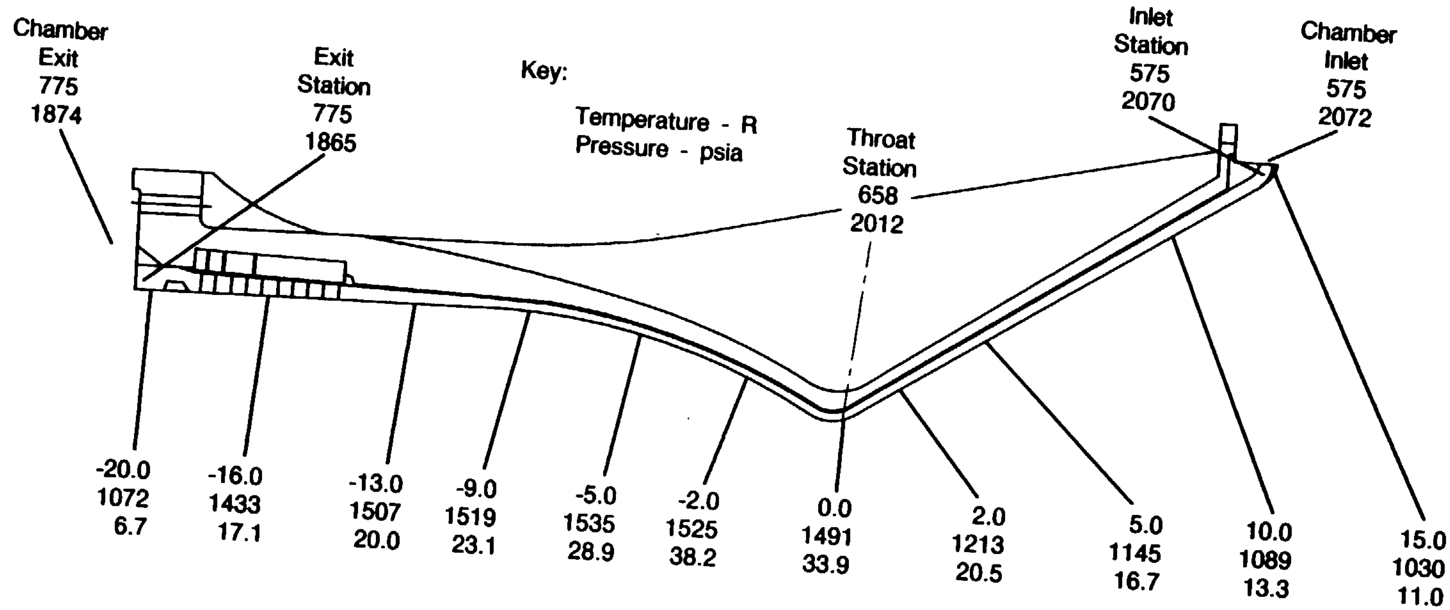
Figure 4.1.4.3-1. STBE LO_2/RP-1 Gas Generator Chamber Design Configuration



Coolant Performance

Thrust = 120%
M_{cool} = 528.0 lbm/ sec

Chamber Heat Transfer Performance

Thrust - lbf	750K	625K
Chamber Pressure - psia	1500	1291
Coolant Flow - lbm/sec	528.0	450.5
Inlet Temperature - R	575.0	564.0
Exit Temperature - R	775.0	775.0
Coolant Heat Pickup - Btu/sec	74143	59270
Inlet Pressure - psia	2073.0	1747.0
Exit Presure - psia	1874.0	1547.0
Pressure Drop - psid	198.0	151.0

Hot Wall Temperature & Heat Flux

Key:

Axial Location - in.
Wall Temperature - R
Heat Flux - Btu/in.^{2} - sec

Figure 4.1.4.3-2. STBE LO_2/RP-1 Gas Generator Chamber Heat Transfer Performance Summary

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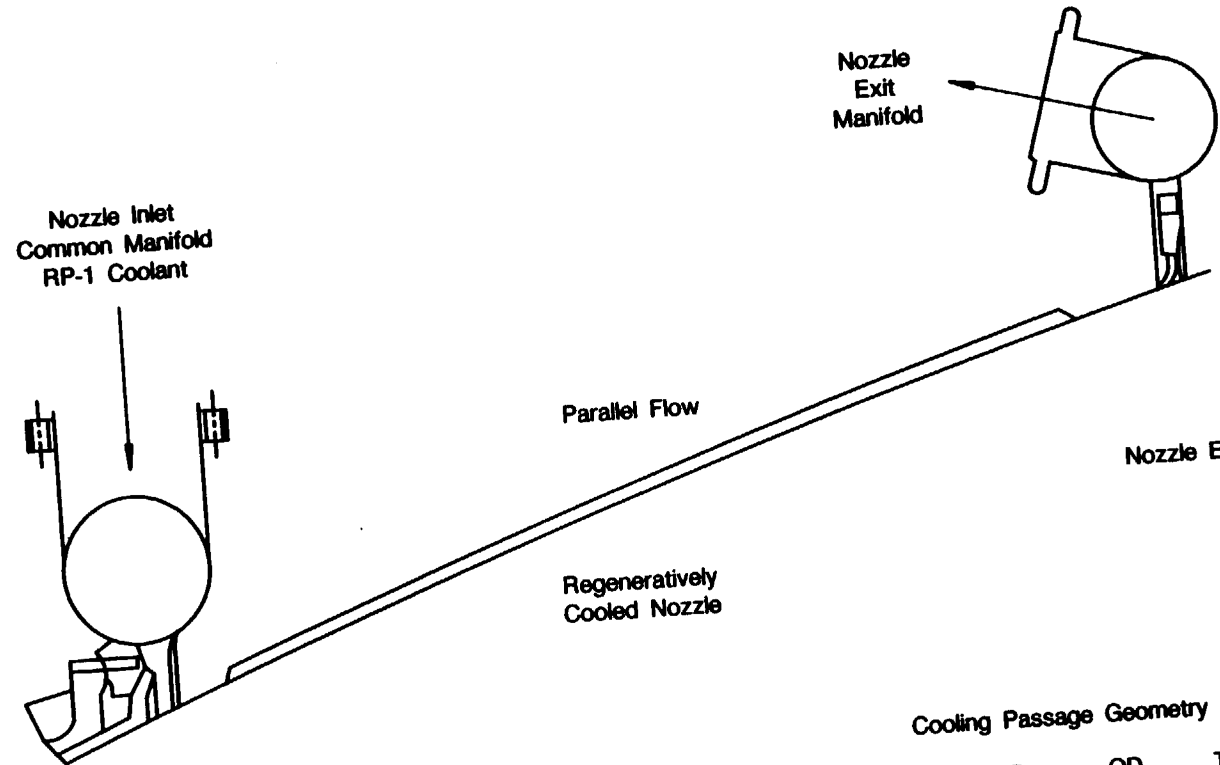

Nozzle Contour Data

Nozzle Length = 20 in.
Inlet Expansion Ratio = 3.28
Exit Expansion Ratio = 7.85
Number of Passages = 630
Nozzle Construction - Super Plastic Inflation Formed
Nozzle Material - AISI 347

Cooling Passage Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Tube Aspect Ratio	Wall Thickness (in.)
15.0	17.91	0.169	0.172	1.020	0.020
18.6	19.87	0.189	0.193	1.021	0.020
23.7	22.51	0.219	0.221	1.024	0.020
28.4	24.98	0.240	0.247	1.026	0.020
35.0	27.74	0.268	0.276	1.029	0.020

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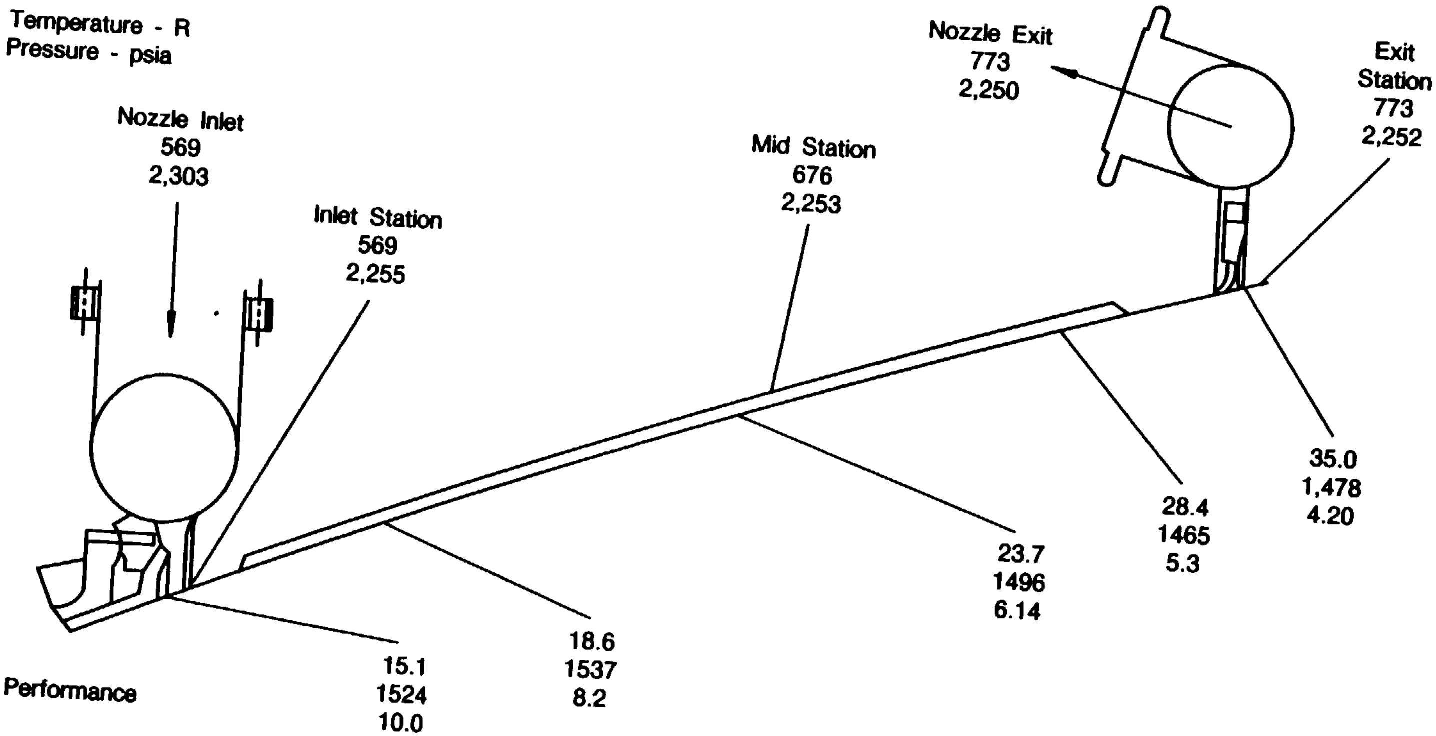
Figure 4.1.4.4-1. STBE LO_2/RP-1 Gas Generator Nozzle Design Configuration



Coolant Performance

Thrust = 120%
M_{cool} = 201.9 lbm/sec

Nozzle Heat Transfer Performance

Thrust - lbf	750K	625K
Chamber Pressure - psia	1500	1291
Coolant Flow - lbm/sec	201.9	172.3
Inlet Temperature - R	569.0	227.0
Exit Temperature - R	773.0	636.0
Coolant Heat Pick-up - Btu/sec	22,982.0	19,200.0
Inlet Pressure - psia	2,303.0	1,941.0
Exit Pressure - psia	2,250.0	1,902.0
Pressure Drop - psid	52.2	39.4

FDA 363341

Figure 4.1.4.4-2. STBE LO_2/RP-1 Gas Generator Nozzle Heat Transfer Performance Summary

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4.1.4.4.2 Film and Radiation Cooled Nozzle

The film and radiation cooled nozzle extends from an expansion area ratio of 7.85:1 to 25:1. Gas generator discharge flow is introduced as a film at the forward end of the radiation nozzle to provide film cooling. The film provides a thermal barrier between the gas path and the nozzle wall, thereby eliminating the need for more complex cooling methods. The highest predicted nozzle wall temperature is 2500 R. Figure 4.1.4.4-3 is a schematic showing wall temperature and heat flux.

4.1.4.5 Engine Configuration and Integration

4.1.4.5.1 GO_2 Heat Exchanger

The GO_2 heat exchanger is designed to provide gaseous oxygen to the oxygen tank for tank pressurization. The GO_2 heat exchanger uses the gas generator exhaust duct as the heat source to vaporize the liquid oxygen as shown in Figure 4.1.4.2-1. The heat exchanger surface is provided by three Haynes 214 stainless steel tubes wrapped in parallel around the gas generator exhaust duct. The gas generator exhaust duct wall is made of beryllium copper with trip strip roughened walls to enhance the heat transfer. The tubes are packed in powdered copper to structurally isolate the tubes from the duct wall, while providing a good heat transfer medium. This heat exchanger will eliminate the possibility of accidental mixing of the oxygen and gas generator exhaust flow, thereby eliminating a category 1 failure mode.

The GO_2 heat exchanger requires three ⅜-inch diameter tubes 61-feet long wrapped around the 12-inch duct. The tubes have 0.015-inch thick walls, and are separated from one another by 0.05 inch, requiring a total duct length of 2.02 feet. Figure 4.1.4.5-1 diagrammatically presents the GO_2 heat exchanger geometry. The GO_2 heat exchanger thermally analyzed for the two STBE engine operating points of 100 percent of 120 percent thrust. The respective oxygen flow rates are predicted to be 5.3 and 6.2 lbm/sec. The heat exchanger is designed to supply 720 R oxygen to the tank. Figure 4.1.4.5-1 also summarizes the predicted heat exchanger performance.

4.1.4.5.2 Engine Performance

The STBE system performance was determined during the preliminary design using the accepted JANNAF methodology. Rigorous procedures have been established for use in calculating chamber/nozzle thrust and specific impulse. The steady-state design point computer simulation provided an initial match of components and definitions of mixture ratio, mass flow, temperature and pressure levels for the detailed performance calculations using the JANNAF methodology. Performance was estimated for both the main chamber flow and the gas generator flow, which is introduced into the main flow through a small nozzle around the perimeter of the main nozzle at an area ratio of 20:1. Table 4.1.4.5-1 lists the detailed performance estimates at the design power level (DPL) of 750,000 pounds sea level thrust while the normal power level of 625,000 pounds sea level thrust is given in Table 4.1.4.5-2. Overall engine performance was calculated by mass weighing the main chamber flow performance with the gas generator flow performance.

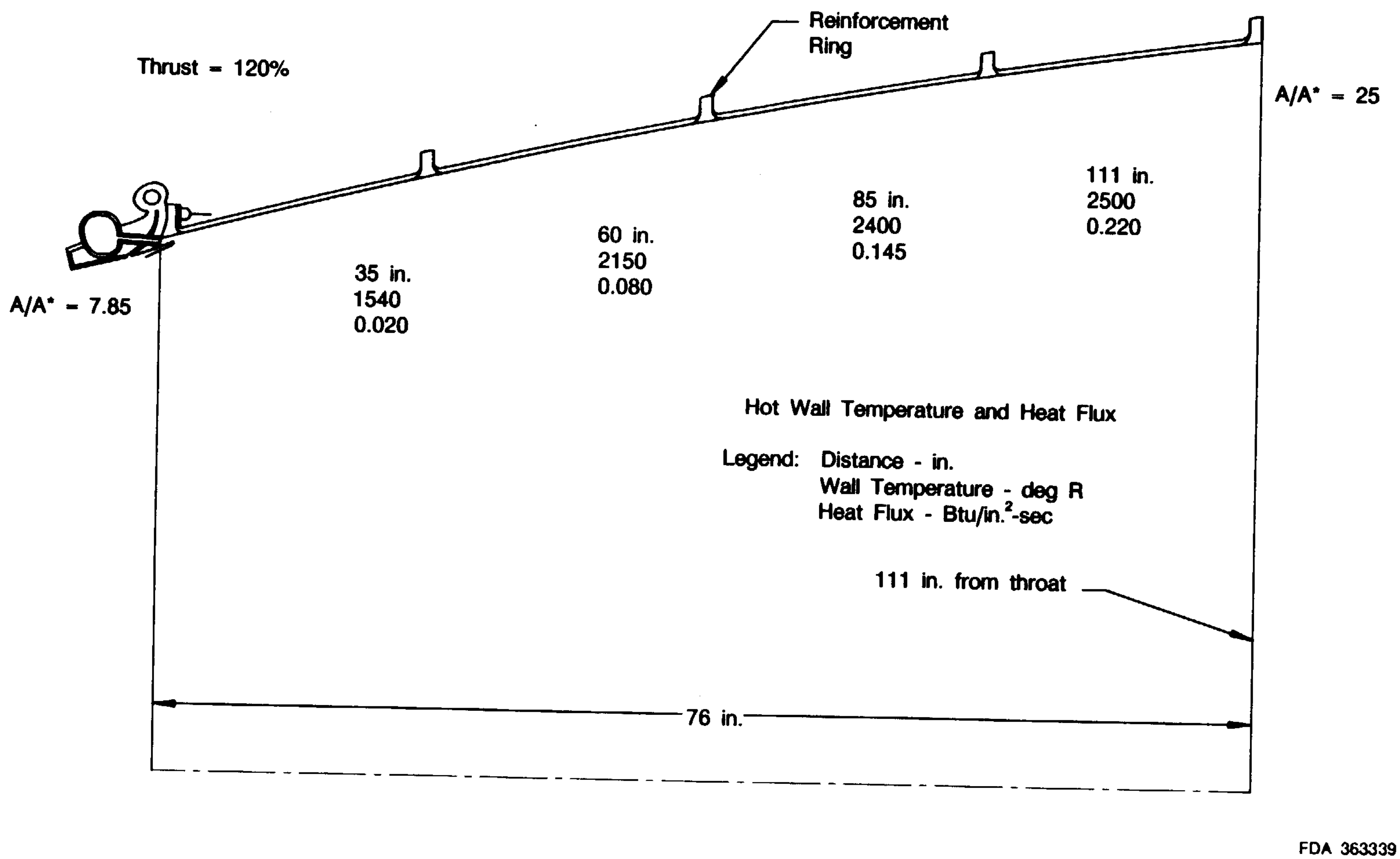

Figure 4.1.4.4-3. STBE LO_2/RP-1 Gas Generator Film and Radiation Cooled Nozzle Thermal Summary

STBE GO_2 Heat Exchanger
Thermal and Flow Performance

Gas Generator Exhaust Duct		
Thrust - %	100	120
Temperature - R	1533	1541
Pressure - psia	120	161
Flow Rate - lbm/sec	71.8	102.7
Oxygen		
Flow Rate - lbm/sec	5.4	6.2
Inlet Temperature - R	172	173
Exit Temperature - R	720	725
Inlet Pressure - psia	1681	1985
Exit Pressure - psia	801	982

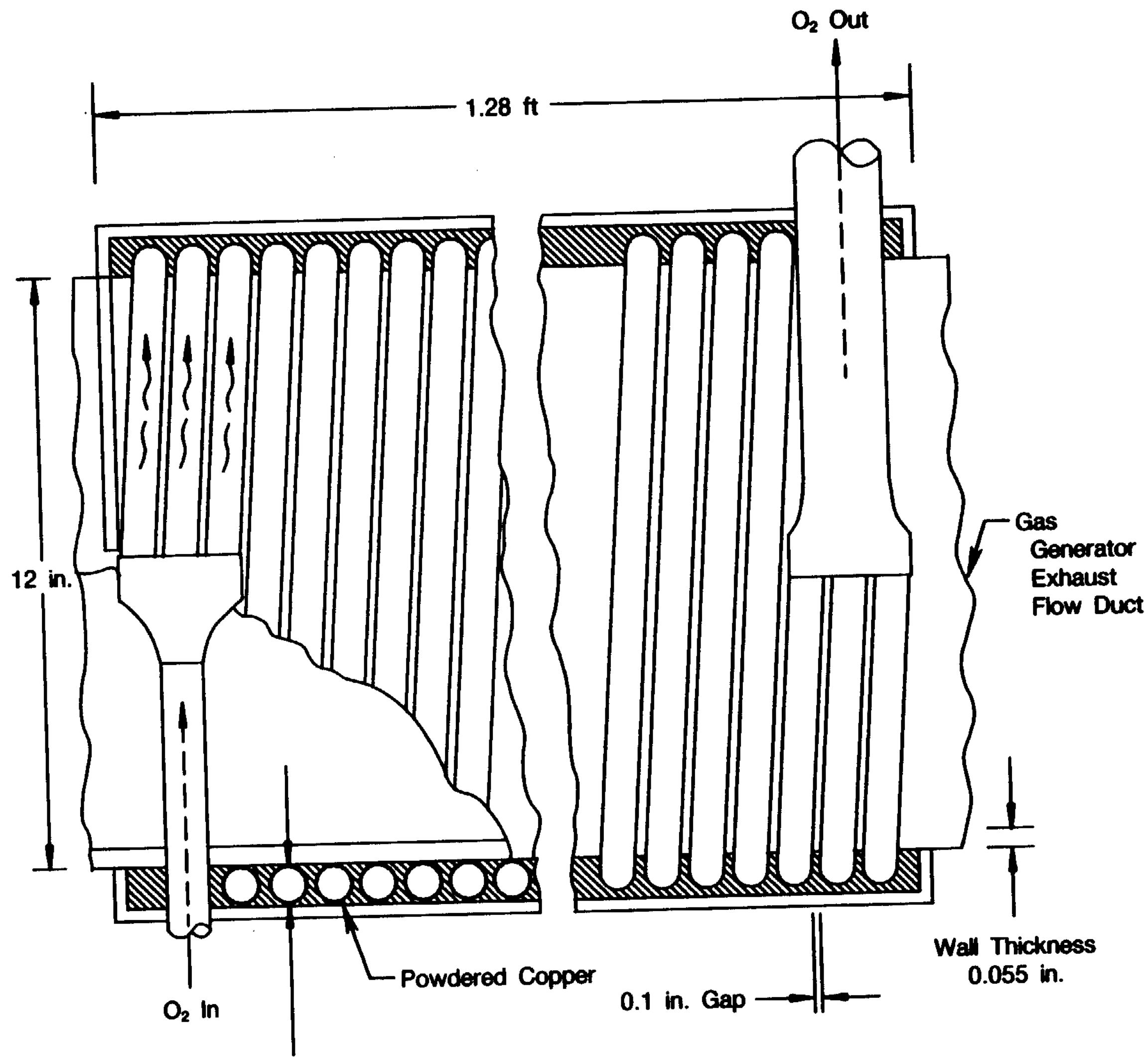

Figure 4.1.4.5-1. STBE LO2/RP-1 Gas Generator GO_2 HEX Geometry and Performance Data

Table 4.1.4.5-1. STME LO_2/RP-1 Performance — Design Power Level

	Main Chamber	*Gas Generator*
Pressure - psia	1501.2	1400.8
Mixture Ratio	3.12	0.114
Area Ratio	25.0	5.0
Ideal I_{sp} - sec	345.4	168.8
ΔI_{sp} ERE - sec	-17.3	0.0
ΔI_{sp} KIN - sec	-0.63	-3.2
ΔI_{sp} TDK - sec	-3.92	-4.5
ΔI_{sp} BLM - sec	-1.47	-3.5
Del. I_{sp} Vac - sec	322.1	157.6
Flowrate - lbm/s	2630.3	101.3
Vac. Thrust - lb	847221.6	15969.5

	Overall Engine
Vac. Thrust - lb	863,191
Vac. Del. I_{sp} - sec	316.0
S.L. Thrust - lb	750,000
S.L. Del. I_{sp} - sec	274.6

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Table 4.1.4.5-2. STME LO_2/RP-1 Performance — Normal Power Level

	Main Chamber	*Gas Generator*
Pressure - psia	1291.9	1191.6
Mixture Ratio	3.04	0.118
Area Ratio	25.0	5.0
Ideal I_{sp} - sec	345.9	141.4
ΔI_{sp} ERE - sec	-17.4	0.0
ΔI_{sp} KIN - sec	-0.76	-3.9
ΔI_{sp} TDK - sec	-3.62	-4.6
ΔI_{sp} BLM - sec	-1.52	-3.2
Del. I_{sp} Vac - sec	322.6	129.7
Flowrate - lbm/s	2259.5	71.3
Vac. Thrust - lb	728946.0	9249.7

	Overall Engine
Vac. Thrust - lb	738,196
Vac. Del. I_{sp} - sec	316.7
S.L. Thrust - lb	625,000
S.L. Del. I_{sp} - sec	268.1

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During this study, detailed aerothermal analyses were conducted to predict component performance levels and these were incorporated into a steady-state computer model of the complete engine. Simplified flow schematics are presented in Figures 4.1.4.5-2 and -3 with key operating parameters noted for each thrust level. Tables 4.1.4.5-3 and -4 define performance of the individual components and their operating environments for the STBE at DPL (120%) and at NPL (100%) respectively.

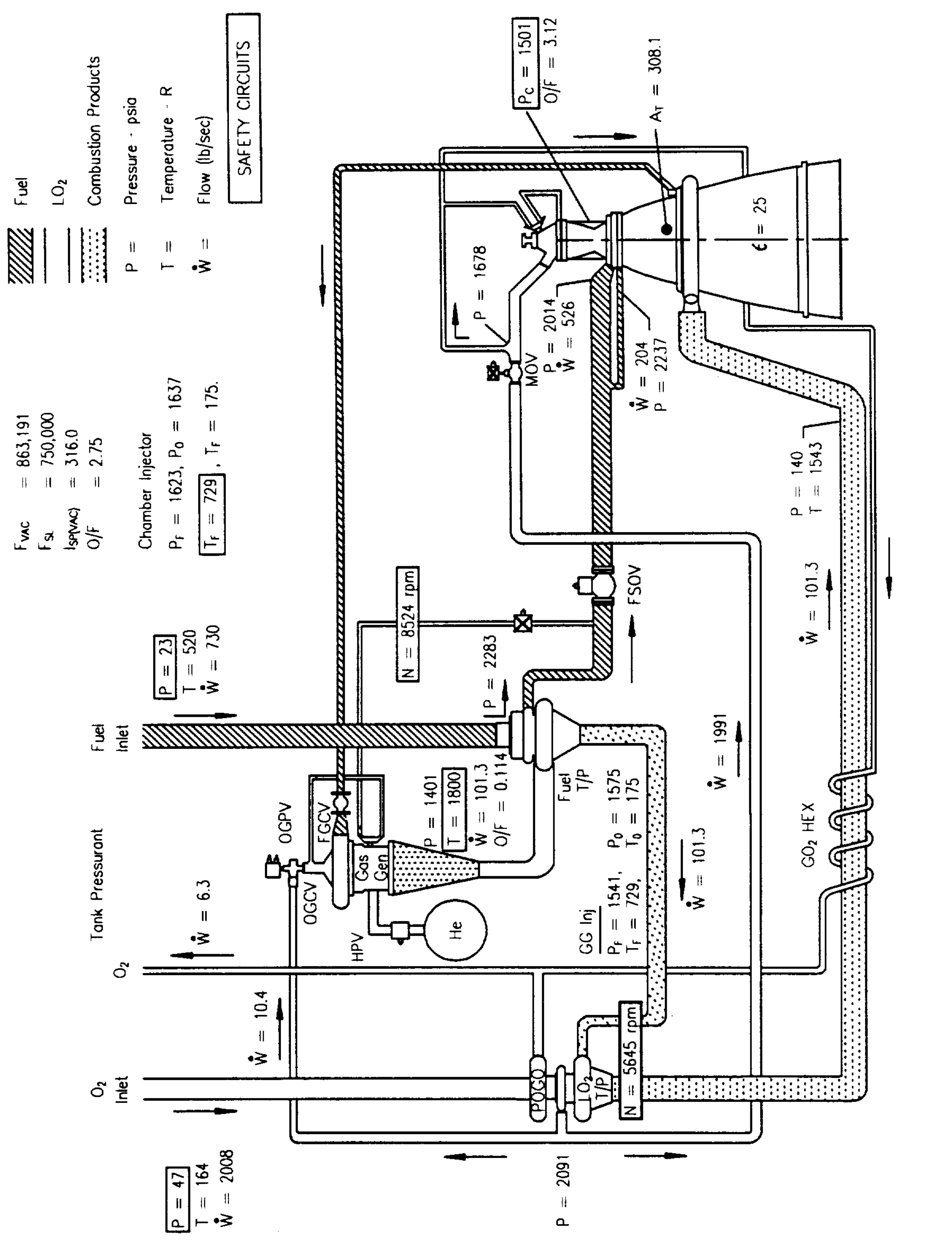

Figure 4.1.4.5-2. STBE LO_2/RP-1 Gas Generator Engine Flow Schematic at Design Power Level (120%)

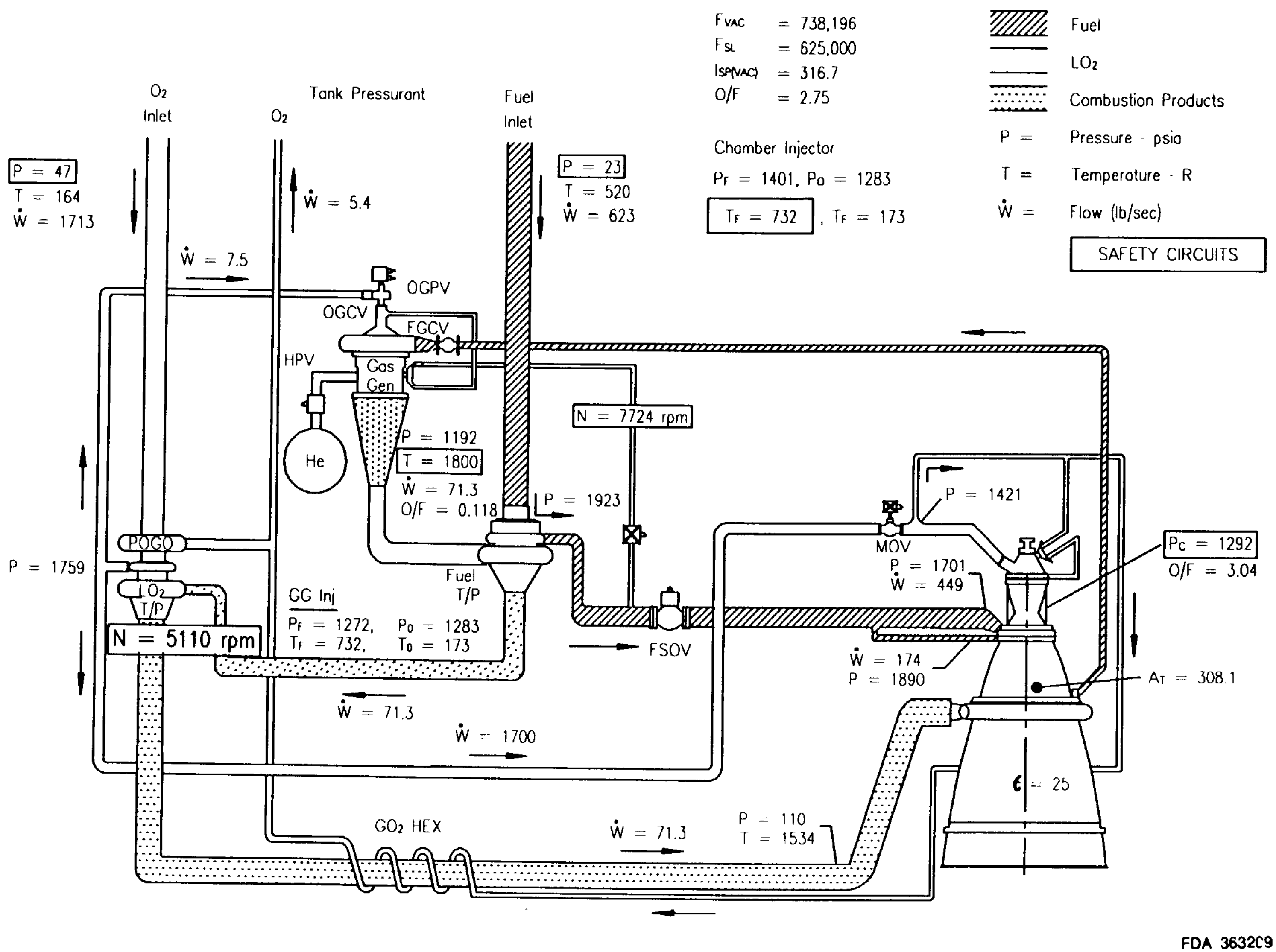

Figure 4.1.4.5-3. STBE LO_2/RP-1 Gas Generator Engine Flow Schematic at Normal Power Level (100%)

Table 4.1.4.5-3. LO_2/RP-1 STBE Gas Generator Performance — Design Power Level

ENGINE PERFORMANCE PARAMETERS

CHAMBER PRESSURE	1501.2
GAS GENERATOR PRESSURE	1400.8
S.L. ENGINE THRUST	750000.
VAC ENGINE THRUST	863191.
DEL. S.L. ISP	274.6
DEL. VAC. ISP	316.0
NOZZLE AREA RATIO	25.0
THROAT AREA	308.1
TC MIXTURE RATIO	3.12
ENGINE MIXTURE RATIO	2.75
CHAMBER COOLANT DP (RP1)	198.
NOZZLE COOLANT DP (RP1)	52.
CHAMBER COOLANT DT (RP1)	160.
NOZZLE COOLANT DT (RP1)	164.
GG MIXTURE RATIO	0.114

ENGINE STATION CONDITIONS

*RP-1 SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	22.8	520.0	729.9	-60.0	55.52
PUMP EXIT	2283.0	563.1	729.9	-49.7	55.08
RP1 CV INLET	2237.4	564.1	525.5	-49.7	55.03
CHM INLET	2013.7	569.2	525.5	-49.7	54.83
CHM EXIT	1815.3	729.2	525.5	63.1	50.44
NOZ INLET	2237.4	564.1	204.4	-49.7	55.03
NOZ EXIT	2185.2	728.4	204.4	62.7	50.62
GG FLOW SPLIT	2184.6	728.4	113.5	62.8	50.62
MFV EXIT	1815.3	729.0	113.5	62.7	50.45
CHM INJ INLET	1623.5	729.2	639.0	63.0	50.45
RP1 GCV INLET	2164.2	728.5	90.9	62.8	50.61
RP1 GCV EXIT	1614.8	729.4	90.9	62.8	50.33
GG INJ INLET	1540.6	729.4	90.9	62.8	51.53

* OXYGEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	164.0	2007.9	61.6	71.17
PUMP EXIT	2091.1	172.9	2007.9	68.4	71.58
O2 TANK PRSZTN	1986.4	173.3	6.288	68.4	71.42
MOV INLET	1986.4	173.3	1991.3	68.4	71.42
MOV EXIT	1677.8	174.4	1991.3	68.4	70.94
CH INJ INLET	1636.7	174.6	1991.3	68.4	70.87
OGCV INLET	1986.3	173.3	10.4	68.4	71.42
OGCV EXIT	1582.8	174.8	10.4	68.4	70.79
GG INJ INLET	1574.9	174.8	10.4	68.4	70.77

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
RP1 MT INLT	1371.0	1800.0	101.3
RP1 MT EXIT	683.0	1700.8	101.3
O2 MT INLT	672.8	1675.3	101.3
O2 MT EXIT	160.0	1543.1	101.3
NOZZLE INLET	140.0		

VALVE, INJECTOR DATA

* VALVES *

	RP1 CV	*RP1 GCV*	* MOV *	* OGCV *	* MFV *
AREA	7.09	4.64	18.85	0.09	1.24
FLOW	525.54	90.90	1991.3	10.39	113.48
DELP	223.8	549.4	308.7	403.5	369.32

	* CHAMBER INJECTORS *		* GAS GEN INJECTORS *	
	* OX INJ *	* RP1 INJ *	* OX INJ *	* RP1 INJ *
AREA	30.39	53.66	0.16	7.37
FLOW	1991.26	639.02	10.39	90.90
DELP	135.60	122.33	174.11	139.46

Table 4.1.4.5-3. *LO_2/RP-1 STBE Gas Generator Performance — Design Power Level (Continued)*

* TURBOMACHINERY PERFORMANCE DATA *

* RP1 TURBINE *		* O2 TURBINE *	
EFFICIENCY	0.77	EFFICIENCY	0.76
HORSEPOWER	10611.	HORSEPOWER	19455.
SPEED (RPM)	8524.	SPEED (RPM)	5645.
DIAMETER	22.45	DIAMETER	33.97
AREA	6.99	AREA	18.91
VEL RATIO	0.38	VEL RATIO	0.40
TIP SPEED	835.64	TIP SPEED	837.37
STAGES	1.	STAGES	2.

* RP1 PUMP *		* O2 PUMP *	
EFFICIENCY	0.74	EFFICIENCY	0.78
HORSEPOWER	10611.	HORSEPOWER	19456.
SPEED (RPM)	8524.	SPEED (RPM)	5645.
S SPEED	972.	S SPEED	1232.
HEAD	5909.	HEAD	4136.
DIAMETER	15.90	DIAMETER	20.62
TIP SPEED	592.	TIP SPEED	508.
VOL. FLOW	5901.	VOL. FLOW	12664.
STAGES	1.	STAGES	1.
HEAD COEF	0.543	HEAD COEF	0.512
FLOW COEF	0.108	FLOW COEF	0.126
SS SPEED	30706.	SS SPEED	28615.

***Table 4.1.4.5-4.** LO$_2$/RP-1 STBE Gas Generator Performance — Normal Power Level*

ENGINE PERFORMANCE PARAMETERS

CHAMBER PRESSURE	1291.9
GAS GENERATOR PRESSURE	1191.6
S.L. ENGINE THRUST	625000.
VAC ENGINE THRUST	738196.
DEL. S.L. ISP	268.1
DEL. VAC. ISP	316.7
NOZZLE AREA RATIO	25.0
THROAT AREA	308.1
TC MIXTURE RATIO	3.04
ENGINE MIXTURE RATIO	2.75
CHAMBER COOLANT DP (RP1)	153.
NOZZLE COOLANT DP (RP1)	40.
CHAMBER COOLANT DT (RP1)	171.
NOZZLE COOLANT DT (RP1)	174.
GG MIXTURE RATIO	0.118

ENGINE STATION CONDITIONS

*RP-1 SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	22.8	520.0	623.0	-60.0	55.52
PUMP EXIT	1923.1	556.2	623.0	-51.3	55.17
RP1 CV INLET	1890.0	556.9	448.5	-51.3	55.14
CHM INLET	1701.0	561.2	448.5	-51.3	54.95
CHM EXIT	1548.0	732.0	448.5	66.2	51.20
NOZ INLET	1890.0	556.9	174.4	-51.3	55.14
NOZ EXIT	1849.7	731.3	174.4	65.9	50.40
GG FLOW SPLIT	1849.3	731.3	110.6	65.9	50.40
MFV EXIT	1548.0	731.8	110.6	65.9	51.20
CHM INJ INLET	1401.3	732.0	559.1	66.2	51.20
RP1 GCV INLET	1839.2	731.3	63.8	65.9	50.40
RP1 GCV EXIT	1308.9	732.2	63.8	65.9	55.62
GG INJ INLET	1272.3	732.2	63.8	65.9	55.57

* OXYGEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	164.0	1713.3	61.6	71.17
PUMP EXIT	1758.5	171.5	1713.3	67.4	71.50
O2 TANK PRSZTN	1682.1	171.8	5.366	67.4	71.38
MOV INLET	1682.1	171.8	1700.4	67.4	71.38
MOV EXIT	1420.6	172.8	1700.4	67.4	70.97
CH INJ INLET	1390.7	172.9	1700.4	67.4	70.92
OGCV INLET	1682.0	171.8	7.5	67.4	71.38
OGCV EXIT	1289.3	173.2	7.5	67.4	70.76
GG INJ INLET	1282.9	173.3	7.5	67.4	70.75

* GAS GEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
RP1 MT INLT	1176.8	1800.0	71.3
RP1 MT EXIT	561.2	1698.5	71.3
O2 MT INLT	552.8	1673.1	71.3
O2 MT EXIT	119.9	1534.3	71.3
NOZZLE INLET	110.0		

VALVE, INJECTOR DATA

* VALVES *

	RP1 CV	*RP1 GCV*	* MOV *	* OGCV *	* MFV *
AREA	6.57	3.72	17.64	0.07	1.34
FLOW	448.53	63.82	1700.4	7.52	110.61
DELP	189.0	530.3	261.4	392.7	301.23

	* CHAMBER INJECTORS *		* GAS GEN INJECTORS *	
	* OX INJ *	* RP1 INJ *	* OX INJ *	* RP1 INJ *
AREA	30.39	53.66	0.14	7.37
FLOW	1700.42	559.14	7.52	63.82
DELP	98.81	109.35	91.26	80.43

Table 4.1.4.5-4. LO_2/RP-1 STBE Gas Generator Performance — Normal Power Level (Continued)

```
***********************************
* TURBOMACHINERY PERFORMANCE DATA *
***********************************
```

```
****************
* RP1 TURBINE *
****************
```

EFFICIENCY	0.73
HORSEPOWER	7635.
SPEED (RPM)	7724.
DIAMETER	22.45
AREA	5.82
VEL RATIO	0.33
TIP SPEED	757.21
STAGES	1.

```
***************
* O2 TURBINE *
***************
```

EFFICIENCY	0.73
HORSEPOWER	13961.
SPEED (RPM)	5110.
DIAMETER	33.97
AREA	17.15
VEL RATIO	0.35
TIP SPEED	757.96
STAGES	2.

```
************
* RP1 PUMP *
************
```

EFFICIENCY	0.74
HORSEPOWER	7635.
SPEED (RPM)	7724.
S SPEED	927.
HEAD	4960.
DIAMETER	15.90
TIP SPEED	536.
VOL. FLOW	5037.
STAGES	1.
HEAD COEF	0.555
FLOW COEF	0.102
SS SPEED	25705.

```
************
* O2 PUMP *
************
```

EFFICIENCY	0.77
HORSEPOWER	13961.
SPEED (RPM)	5110.
S SPEED	1177.
HEAD	3463.
DIAMETER	20.62
TIP SPEED	460.
VOL. FLOW	10806.
STAGES	1.
HEAD COEF	0.523
FLOW COEF	0.118
SS SPEED	23925.

4.1.4.5.3 Engine Costs

This section summarizes cost estimates for the 750K SL thrust, 1500 psia chamber pressure, RP-1/LO_2 STBE Gas Generator cycle. Table 4.1.4.5-5 summarizes significant costs for the engine.

Table 4.1.4.5-5. *RP-1/LO_2 STBE Gas Generator Costs*

Total Development Cost (DDT&E), M$1500*
Production Cost (TFU), M$11.9
Operations Cost/Launch/Engine, M$0.200**
Constant FY87$

*Applies to developing a stand-alone booster engine configuration.
**Based on the 100th mission, 10 missions per year, and seven boosters per vehicle.

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The DDT&E Cost includes all of the functions required to design, develop, test and evaluate the engine system. All of the DDT&E functions shown in the ALS engine WBS (see Volume III) have been included. Development Cost is based on a 90-month phase C/D program with 960 engine firings for the RP-1/LO_2 STBE. Sufficient accountable firings have been included in the program to demonstrate 0.99 engine reliability with one failure.

The engine Theoretical First Unit (TFU) production cost includes all the recurring operational production cost elements specified in the ALS engine WBS. It includes manufacturing and acceptance of the Integrated Engine System, System Engineering and Integration, Program Management, Facilities Maintenance and Tooling Maintenance. The TFU estimate is based on a lot size of 100 and a 90-percent learning curve.

The Operations Cost per launch per engine includes all costs associated with the operational flight program as described in the ALS engine WBS. It includes Program Management, System Engineering and Integration, Facilities Maintenance, Operation and Support, and Training. The Operations Cost is based on a flight rate of 10 missions per year and it is the estimated cost that will be achieved after 100 total missions have been flown.

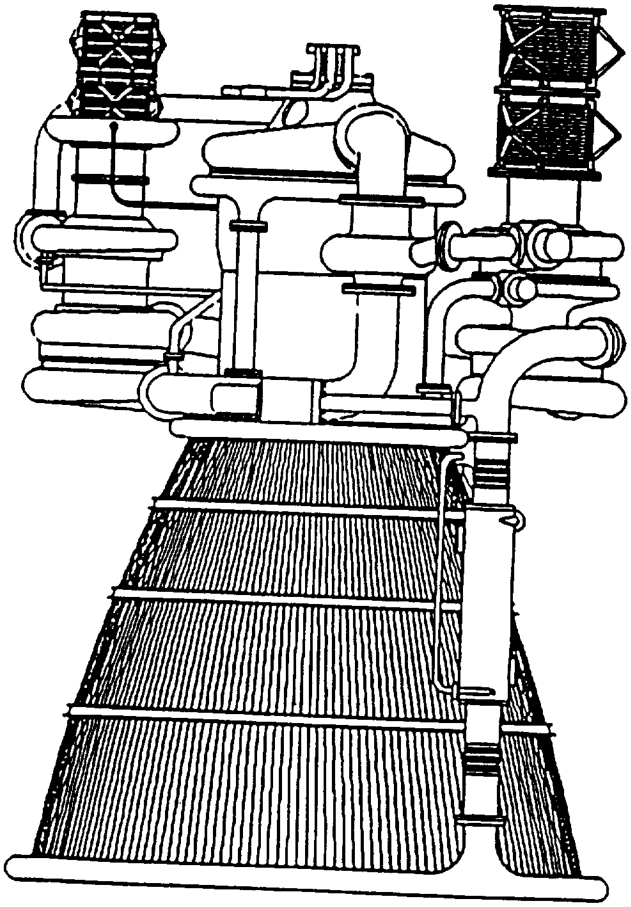
4.2 SPLIT EXPANDER CYCLE ENGINES

4.2.1 Derivative Split Expander Cycle Engine

4.2.1.1 Engine Design Evolution

The derivative, or modified split expander cycle engine study conceptual design was initiated as a result of the emerging need for a booster engine derived from the main engine. The 580K lbf vacuum thrust split expander main engine is designed for unique application to a core vehicle and delivers 433.9 seconds of vacuum specific impulse at the design power level using LO_2/H_2 as propellants. By utilizing as much hardware as possible, a derivative of this engine is designed to power a booster vehicle using LO_2/CH_4 as propellants. Both engines are presented in Figure 4.2.1.1-1 with overall engine characteristics.

The derivative engine studies conducted during the last quarter of 1988 and first quarter of 1989 showed that maximum part commonality to the unique STME Split Expander engine could be achieved only at low booster engine thrust levels in the 300-400K range. Since the minimum acceptable sea level thrust for a booster engine application is 600K lbf, several new components were designed for the booster engine. The derivative split expander design hardware common to the STME Split Expander is shown in Table 4.2.1.1-1. Detailed discussion of this engine is presented in the following paragraphs.



Propellants	H_2/LO_2	CH_4/LO_2
Mixture Ratio	6.0	3.5
Chamber Pressure - psia	896	734
Thrust - Vacuum - sec	580,000	706,500
- Sea Level - sec	436,187	600,032
Specific Impulse - Vacuum - sec	433.9	327.7
- Sea Level - sec	326.3	278.3
Nozzle Area Ratio	28	13.5
Diameter - in.	116	104
Length - in.	187	140
Weight - lb	5,084	6,193

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Figure 4.2.1.1-1. STBE Derivative Split Expander Engines at Design Conditions

Table 4.2.1.1-1. *STME and Derivative STBE Split Expander Engines Common Hardware Components*

Turbomachinery

- Fuel Pump Housing Flow Paths
- Fuel Pump Impeller Flow Path
- Ball and Roller Bearings
- Turbine Outer Seals
- Tiebolt Shaft and Disks (Modified Blade Attachments)
- Internal Labyrinth Seals
- Major Flange Seals
- Bolts, Nuts, Studs, Washers, Pins
- 1st and 2nd-Stage Impeller Castings
- Uniform Cross Section Static Housing Seals
- Inducer Retaining Bolts
- Blade Retaining Rings, Tip Seals
- Spacers, Bearing Sleeves, Wave Washers Made from Same Forging or Identical Hardware

Engine Controls

- Engine Controller
- Engine and Component Instrumentation

Combustion Devices

- Igniter Assembly — Main Injector

Engine Assembly

- Ducting
  50% Small Lines
  20% Large Lines
- Engine Vehicle Interface Points
- GO_2 HEX
- POGO Suppressor
- Fuel Inlet Flex Joints
- Fasteners, Seals

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4.2.1.2 Engine Cycle

The derivative STBE is a split expander cycle with liquid oxygen and liquid methane as the propellants. It is a derivative of the STME LO_2/hydrogen engine, and is intended to utilize as many STME hardware components as possible. This engine operates at a main chamber pressure of 734 psia at a fixed thrust level (NPL) of 600K lbf. The nozzle area ratio is optimized, for a booster engine application, at 13.5:1 and results in a delivered sea level specific impulse of 328 seconds.

4.2.1.2.1 Flow Path Description

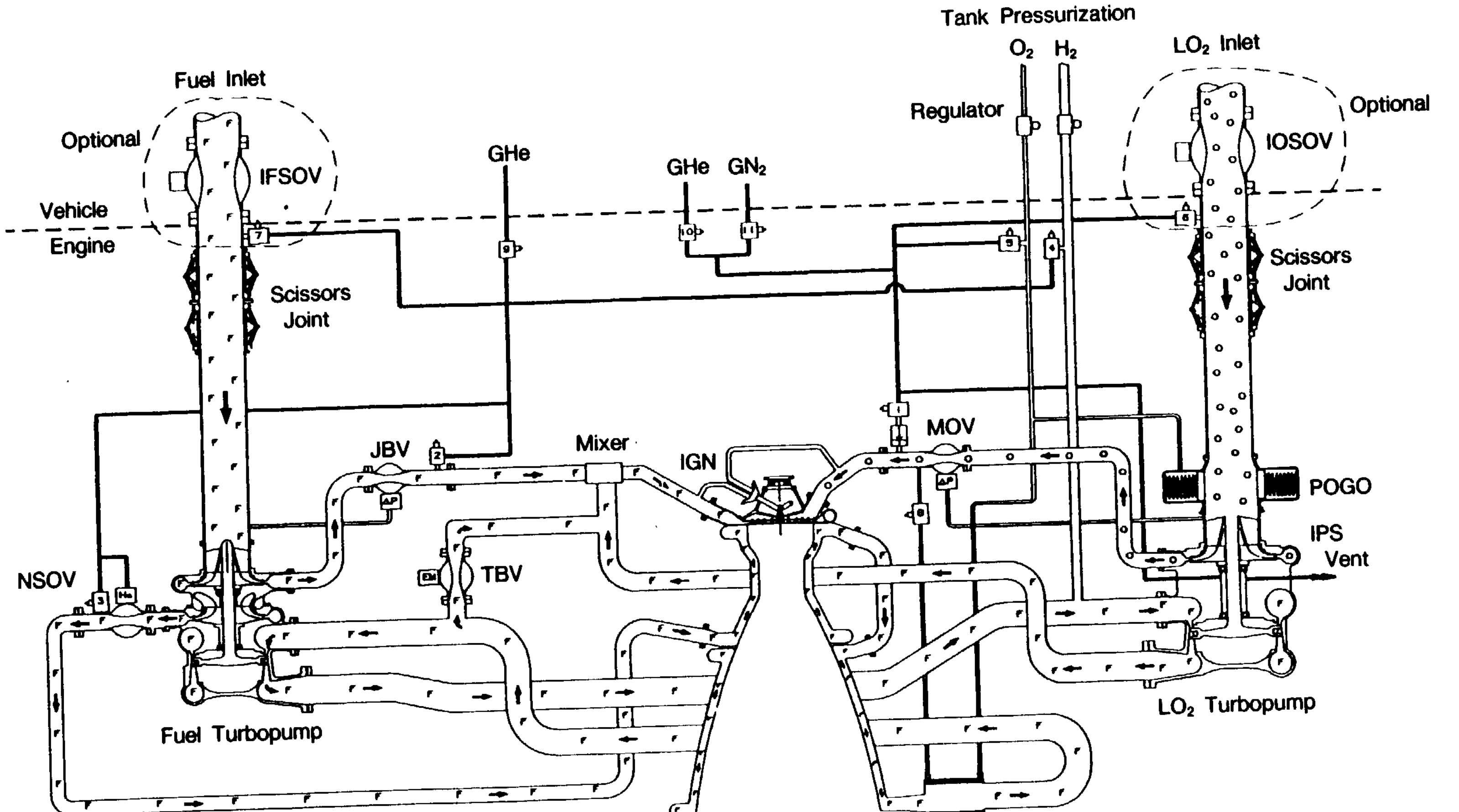
A simplified flow schematic for the derivative STBE showing only the major flow paths and components is presented in Figure 4.2.1.2-1. Liquid oxygen and methane enter the engine at a NPSH level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps. No boost pumps are required in these systems. At normal power level, the methane pump operates at 10953 rpm to provide a first-stage pump discharge pressure level of 2546 psia. From the first-stage pump exit, 57 percent of the flow is routed to the second stage of the methane pump. The second-stage pump discharge level is 5740 psia. From the second-stage pump exit, the methane is routed through the nozzle shutoff valve into a split manifold chamber/nozzle. This heated methane is then used to provide power to drive the propellant pumps. Ninety percent of the nozzle cooling flow is routed through the turbines. The warm (689 R) methane gas is initially expanded through the methane pump drive turbine and is subsequently routed to a second turbine that powers the oxygen pump. The turbine exhaust is then routed to a mixer, where it combines with the remainder of the methane flow, and is then injected into the main chamber. At normal power level, the oxidizer pump operates at 5014 rpm to provide an oxygen pressure level of 1224 psia. From the pump exit, the oxygen flow is routed through a control valve and injected directly into the main chamber. Some key operating conditions for the pumps are listed in Table 4.2.1.2-1.

4.2.1.2.2 Engine Operation

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor or on the pad, because all fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the LO_2 injector is primed prior to opening the fuel shutoff valve to assure liquid oxidizer flow. During the start and shutdown, a small helium purge is used in the main chamber injector to eliminate the danger of hot gas flow reversals during transient operation. Main chamber ignition will be accomplished with an electrical, spark-excited, oxygen/methane torch igniter.

Engine operation is controlled by a timed sequence of the five valves: nozzle shut-off valve, (NSOV), jacket bypass valve, (JBV), fuel shut-off valve, (FSOV), turbine shut-off valve, (TBV), and main oxidizer valve (MOV) (Figure 4.2.1.2-1). Engine acceleration is accomplished by scheduling the valves on open-loop schedules to full thrust.



Sol 1 (N/C) - MOV Purge Solenoid
Sol 2 (N/C) - JBV Purge Solenoid
Sol 3 (N/C) - CSOV Purge Solenoid
Sol 4 (N/C) - Fuel Pressurization Line Purge
Sol 5 (N/C) - LO_2 Pressurization Line Purge
Sol 6 (N/C) - LO_2 System Purge Solenoid
Sol 7 (N/O) - Fuel System Purge Solenoid
Sol 8 (N/C) - Antiflood Solenoid
Sol 9 (N/C) - GHe Purge (Fuel)
Sol 10 (N/C) - GHe Purge (Oxidizer)
Sol 11 (N/C) - GN_2 Purge

NSOV Nozzle Shutoff Valve
MOV Main Oxidizer Valve
JBV Jacket Bypass Valve
TBV Turbine Bypass Valve

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Figure 4.2.1.2-1. Simplified Flow Schematic for STBE Derivative Split Expander Cycle Engine

Table 4.2.1.2-1. Derivative STBE Split Expander Operating Conditions

Fuel Turbopump	
Pressure — psia	5,740
Maximum speed — rpm	10,953
Turbine Temperature — R	877
Pump Stages	2
Turbine Stages	2
Oxidizer Turbopump	
Pressure — psia	1,224
Maximum Speed — rpm	5,014
Turbine Temperature — R	719
Pump Stages	1
Turbine Stages	2
Main Chamber	
Chamber Pressure — psia	734
Heat Pickup — Btu/sec	138,510
Coolant Flow — lbm/sec	276

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During preconditioning, all of the valves are closed except for the MOV; which is approximately 25 percent open for simultaneous LO_2 injector cooldown. Once the engine is adequately preconditioned, the MOV opens further to completely fill the LO_2 injector prior to ignition. During the process of filling the injector, the NSOV remains closed to prevent cooling of the nozzle/chamber cooling jacket. Once the LO_2 injector is full, the NSOV and the FSOV are opened so the fuel can flow freely to the injector. At this point, the JBV and the TBV remain closed to force all of the available fuel through the turbines. After ignition and upon sufficient power from the turbines, the JBV opens to bypass flow from the pump first-stage discharge to the mixer. Once the desired thrust level is achieved, the TBV opens to control turbine power. At this point, the engine should be at its steady-state conditions.

Engine shutdown is accomplished using a time based scheduling of the propellant valves. First, the TBV is further opened to reduce turbine power and slow the pumps. Then the methane system is shut down by closing the JBV, NSOV and FSOV in that order to purge the fuel system of excess methane. Finally, the oxidizer system is shut down by closing the MOV.

4.2.1.3 Turbomachinery

4.2.1.3.1 Oxidizer Turbopump Hardware Description

The oxidizer turbopump is configured as a single centrifugal shrouded turbopump with an inlet inducer driven by a two-stage axial flow turbine. The inducer and impeller, made of fine grained cast Inconel 718, are coupled to the turbine through a shaft made of super A-286 material to a cast aluminum integrally bladed turbine disk. The pump inlet/discharge housing is made of AISI 34755T while the remaining housings are made from aluminum. The ball and roller bearings, made of 400C material, will be used to support the pump rotor system. The oxidizer turbopump and its major components are shown in Figure 4.2.1.3-1.

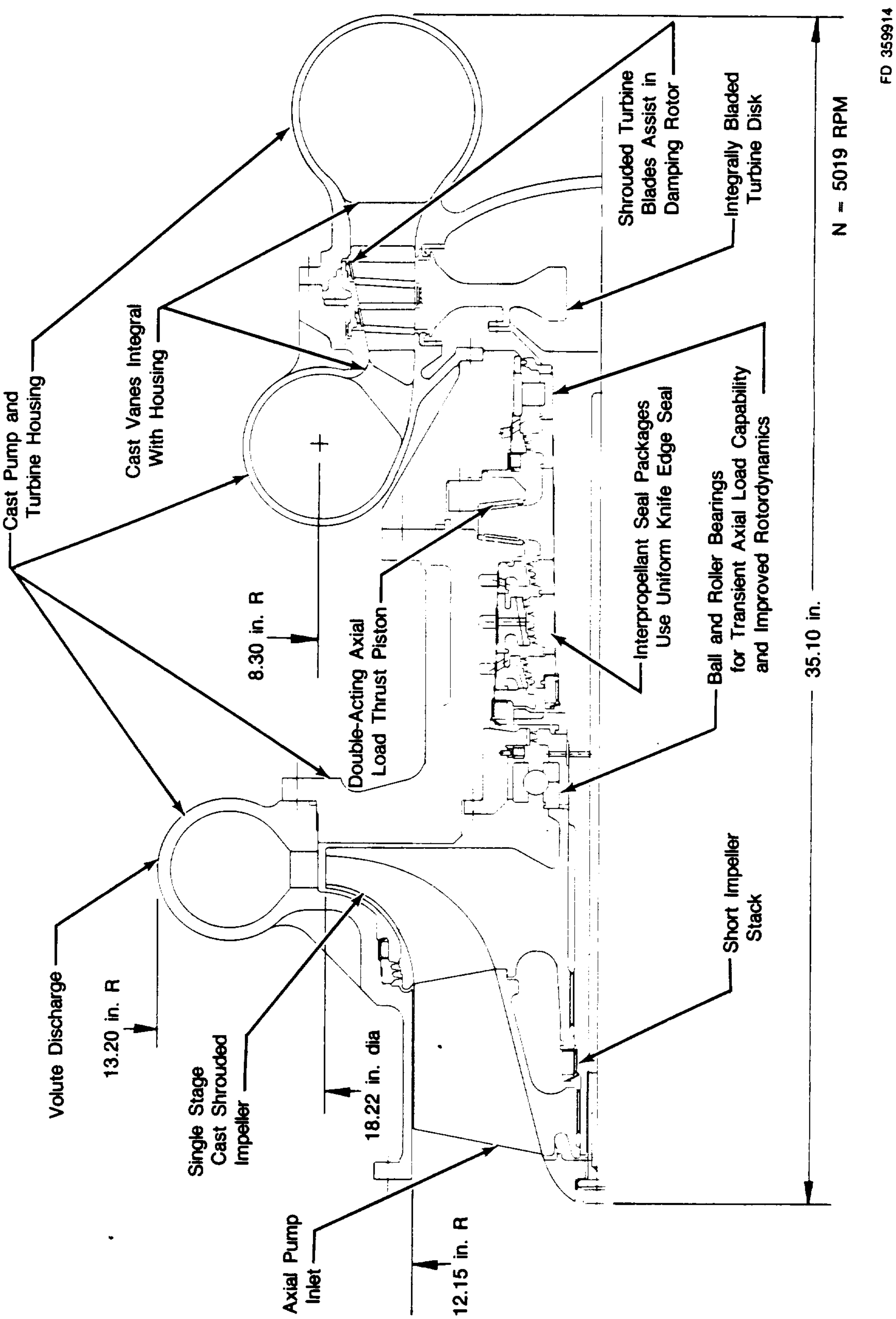

Figure 4.2.1.3-1. STBE Derivative Split Expander Oxidizer Turbopump

The rotor thrust balance system is accomplished through the incorporation of a double acting thrust balance system on the turbine side of the interpropellant seal turbopump in a liquid fuel environment so as to avoid any rub in a LO_2 environment. Externally supplied high pressure fuel is used for thrust piston actuation and for roller bearing and turbine coolant. The rotating thrust piston is made of forged Inconel 718 and its mating surface of the stationary housing is an insert made of Bearium B-10 material (leaded bronze). Axial travel of the rotor is controlled at this location.

The double-acting thrust piston provides thrust balance capability to the rotor system by controlling axial imbalance loads during startup, steady-state, and shutdown operation. As an axial imbalance load occurs, the rotor moves axially, which opens an orifice that supplies high-pressure fuel to the side of the piston in which the rotor has traveled. At the same time, the opposite piston face is now vented to low pressure fuel, resulting in a reaction thrust load that restores the rotor to its initial position.

While the roller bearing is cooled by fuel, the ball bearing is cooled with LO_2. Oxidizer flow along the backside of the impeller is used as bearing coolant; then it is recirculated to the inducer inlet through a controlling orifice/hole in the bearing carrier and shaft.

The interpropellant seal package employs a labyrinth seal design with a helium buffer cavity. This design is similar to the SSME ATD LO_2 turbopump design. An oxidizer-side vaporizer is incorporated to reduce the overboard leakage.

The turbine inlet housing is a cast volute integrating the first-stage turbine vane, and contains the placement of the turbine tip seal lands. Attachment of the inlet housing to the pump housing is achieved with a flexible arm designed to provide thermal compatibility between the two housings.

4.2.1.3.2 Fuel Turbopump Hardware Description

The fuel turbopump is configured as an inlet inducer with a two-stage centrifugal impeller pump driven by a two-stage axial flow turbine. The inducer, made of aluminum, and the impellers, made of fine grained titanium A-110 ELI, are coupled to the turbine through an Inconel 718 shaft to a forged aluminum bladed turbine disk with brazed blade tip shrouds. The pump and turbine housings are fabricated of cast aluminum to minimize machining costs. One ball and roller bearing, made of 440C material, support the pump rotor system. Figure 4.2.1.3-2 shows the fuel turbopump and its major components.

The rotor thrust balance system is accomplished by incorporating the thrust piston into the second-stage impeller. A double acting, double orifice thrust piston has been configured into the front and back side of the impeller. The thrust piston is designed to control axial imbalance loads during engine startup, steady-state, and shutdown conditions. As the thrust imbalance load occurs, the rotor moves axially, which then opens an orifice at the impeller tip, introduces pump discharge high pressure fuel to the side of the impeller in which the rotor has traveled. At the same time, the opposite impeller face is vented to low pressure fuel, resulting in a reaction thrust load to restore the rotor axial position. Both sides of the thrust piston are fed with second-stage impeller discharge pressure. Axial travel is limited by a forward stop on the impeller ID shroud face and by an aft stop on the ID of the impeller back face.

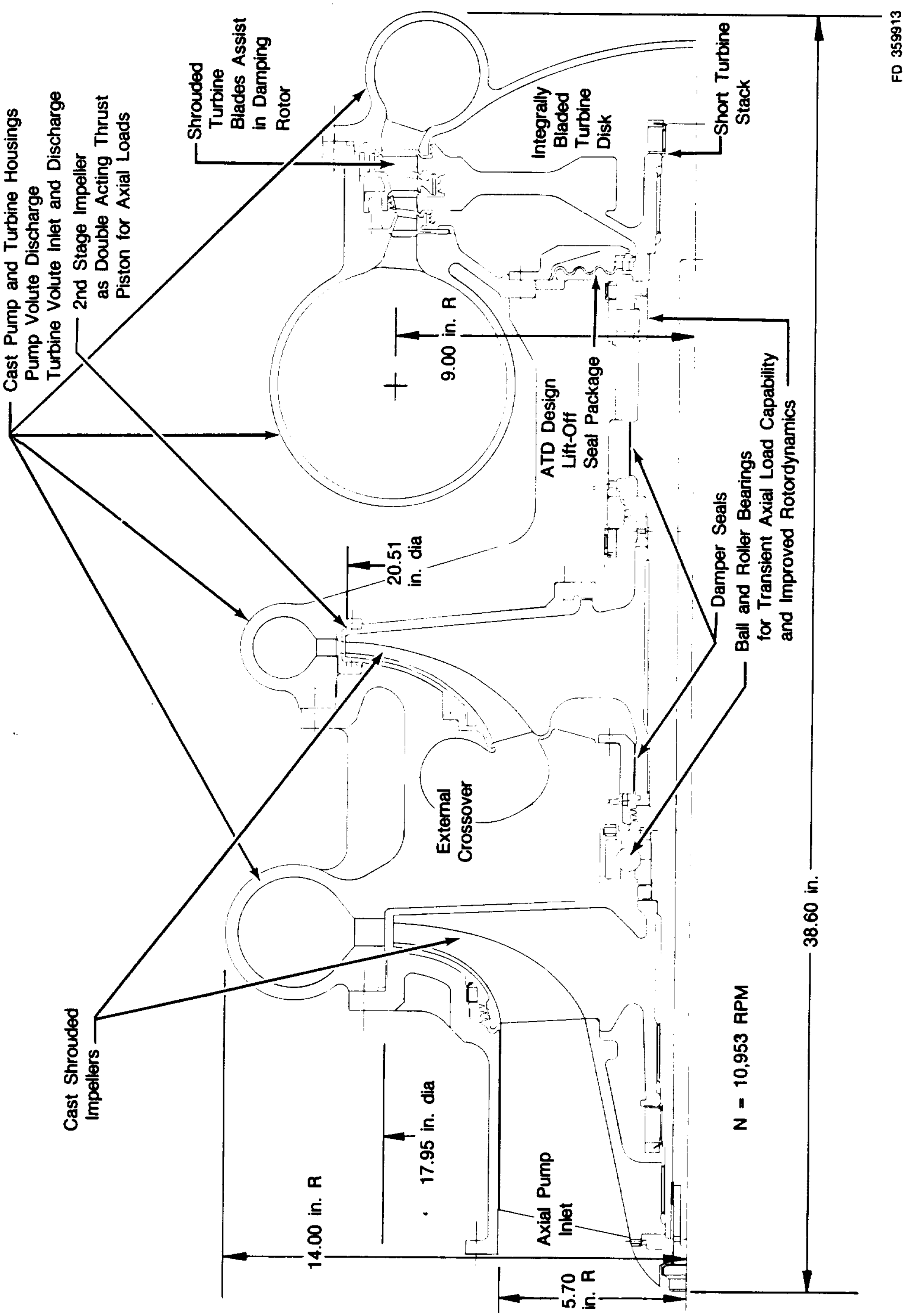

Figure 4.2.1.3-2. STBE Derivative Split Expander Fuel Turbopump

The ball and roller bearings are similar to the bearings used on the SSME-ATD fuel turbopump. The ball bearing is cooled by first-stage discharge pressure bled off the impeller back face and flow controlled by the labyrinth seals near the outer diameter of the impeller. The roller bearing is cooled by second-stage discharge pressure that is supplied to the bearing via internal passages through the pump housing. Roller bearing coolant is then discharged into the turbine disk cavity to be used as turbine coolant.

A diaphragm type lift-off seal (similar to the ATD fuel turbopump) is incorporated in the fuel pump design to prevent cooldown flow from entering the turbine during the pre-start sequence. At engine start, pump pressure increases so that the lift-off seal is deflected and flow is permitted through the bearing and into the turbine for additional turbine cooling requirements.

4.2.1.3.3 Turbomachinery Rotordynamics

The P&W Advanced Launch Systems (ALS) Program is designed to produce reliable, low-cost rocket engine turbopumps. Pratt & Whitney uses proven design criteria and analytical methods in the design of rotordynamic operation for jet engine rotors and rocket engine turbopumps. Each Derivative Split Expander Oxygen Turbopump (DSEOT) and Fuel Turbo-pump (DSEFT) design incorporates configuration changes which result in stiffer rotors, bearings, and rotor support structures with the addition of roughened stator damper seals. For optimum rotordynamics, each rotor is supported by strategically located, stiff, durable bearings. These changes result in a significant improvement to the first fundamental bending mode of the rotor, moving it well beyond the maximum operating design speed. This, in addition to an improved rotor balance procedure, results in an effective low-speed balance of the rotor for low synchronous response. Rotor stability in the DSEOT and the DSEFT have been improved by designing the turbopumps to operate below the first vibrational mode of the rotor. Increased stability margin in each turbopump is provided by the introduction of roughened stator damper seals into the design.

4.2.1.3.3.1 Oxidizer Turbopump Rotordynamics

The primary goals in the design of the DSEOT have been to provide: (1) greater than 20 percent margin over design speed for the fundamental first bending mode for low synchronous response, (2) a sufficient stability margin, and (3) a high integrity rotor balance. Meeting these provisions has required optimization of the mechanical design of the rotor, bearings, rotor support, damper seals, and housings for successful rotordynamic characteristics. The initial P&W DSEOT design moved the first fundamental rotor bending mode, with high strain energy, to well above the design speed, effectively eliminating the synchronous response due to rotor imbalance. The pitch and bounce modes of the rotor occur at 115 and 563 percent of operating speed. These modes are classified as rigid body modes and are of relatively low rotor strain energy content. They are shown in Figures 4.2.1.3-3 and -4.

Rotor bearing stiffness plays an important role in the dynamic behavior of all turboma-chinery. In high-pressure rocket turbopump designs, P&W realizes the need for the combined rotor support system (i.e., bearing, carrier, and backup structure) to approach or exceed the relative stiffness of the rotor structure.

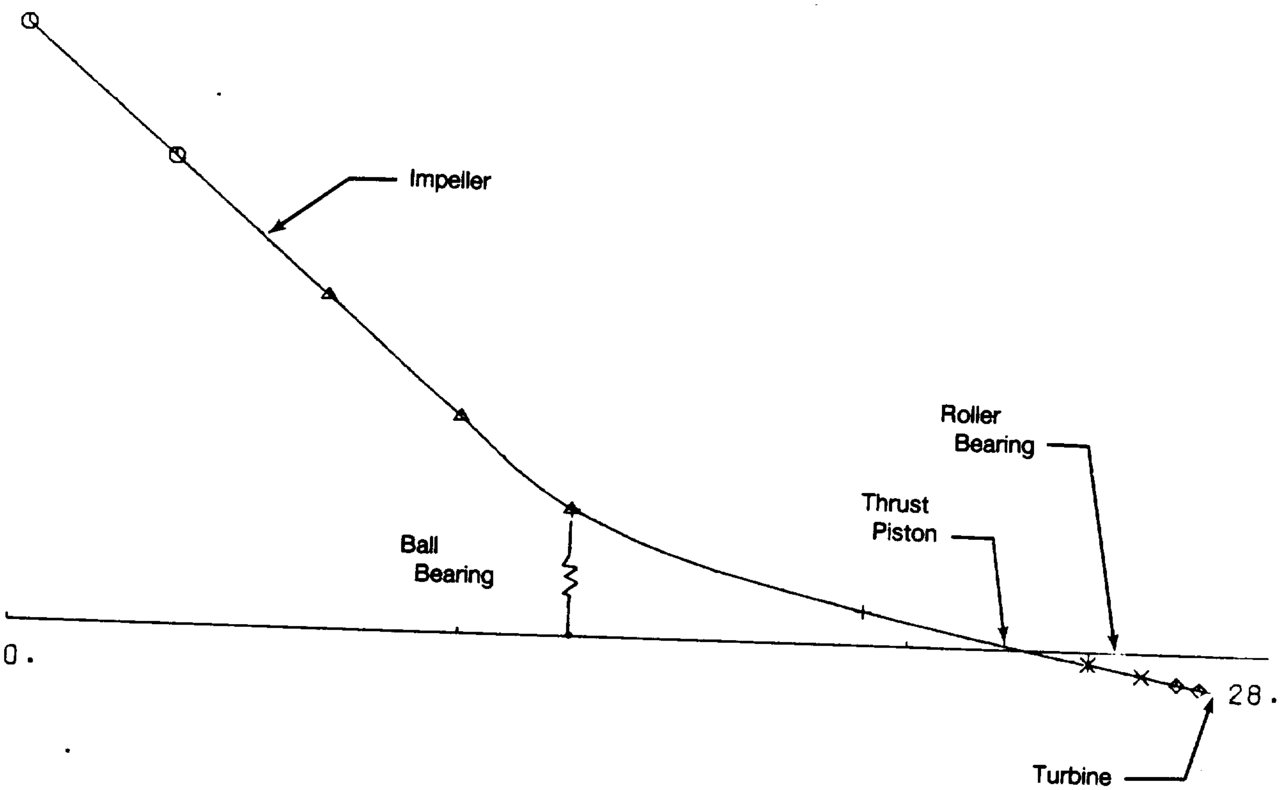

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Figure 4.2.1.3-3. STBE Derivative Split Expander LO_2 Turbopump Critical Speeds Analysis Showing the Pump Pitch Mode of the Rotor at 115% Design Speed (RPM = 7615)

The rotor critical speed analysis has been used to set initial design requirements for each bearing stiffness. The pump end bearing is a large diameter, high load capacity ball bearing with minimal internal radial clearance (IRC) and deadband. The turbine end bearing is a large diameter, high load capacity roller bearing with a negative IRC. High rotor stiffness, coupled with stiff rotor design, without exceeding successful P&W bearing DN experience, ensure that successful rotordynamics criteria are met for this application.

Rotordynamic stability analysis will be used as a design tool to determine the final damper seal configuration requirements for optimized system dynamics. However, the DSEOT is designed such that damper seals are not critical to the dynamics of the rotor system. Each of the seals is designed for high damping, moderate stiffness, and minimal leakage. The incorporation of damper seals into the turbopump design provides: (1) reduced synchronous response throughout the operating speed range resulting in lower dynamic bearing loads and rotor deflections, (2) increased margin on the onset speed of instability (OSI), and (3) additional rotor load support between the bearings. Locations for the damper seals are being investigated to be incorporated into the next phase of design.

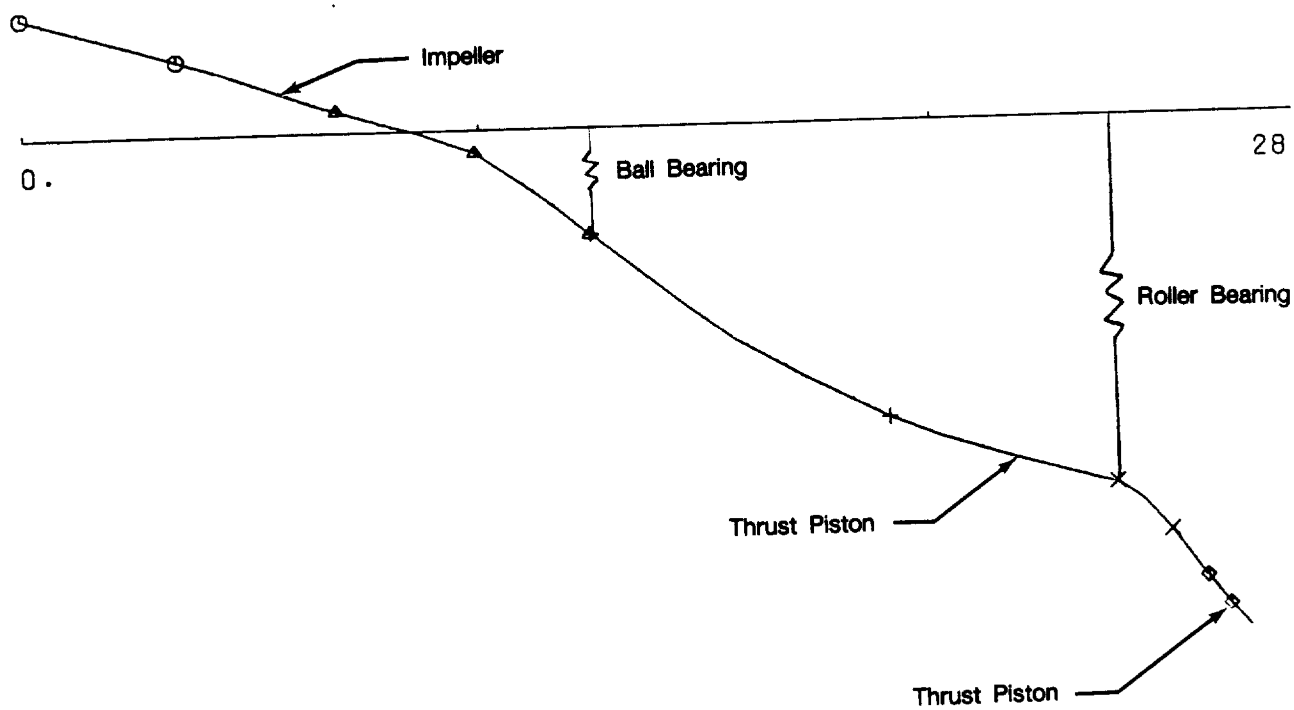

Figure 4.2.1.3-4. STBE Derivative Split Expander LO_2 Turbopump Critical Speeds Analysis Showing the Turbine Bounce Mode of the Rotor at 563% Design Speed (RPM = 37196)

The DSEOT design provides for improved rotor balancing. Each major rotating component will be double piloted and indexed to the through tiebolt for positive concentricity control and balance repeatability. In addition, each major rotating component will be dynamic check balanced in detail to provide minimal residual force and moment imbalance. The dynamic balance of the rotor assembly will be completed with corrections in two planes.

The DSEOT design proposed by P&W results in acceptable rotordynamic characteristics throughout the operating range. The lightweight stiff rotor and bearing design have reduced the synchronous response due to the first fundamental rotor bending mode having been driven to more than 779 percent above the design speed. Thus, the stability of the rotor is significantly improved by avoiding the subsynchronous excitation associated with the critical speeds below 50 percent of the design speed.

A critical speed summary for the DSEOT is provided below.

W_{cr} (rpm)	% Design Speed	% Rotor Strain Energy	Mode Description
7615	115	35.0	Pump Pitch
37196	563	12.4	Turbine Bounce
58056	879	85.0	1st Bending

4.2.1.3.3.2 Fuel Turbopump Rotordynamics

The three primary goals in the design of the DSEFT have been to provide: (1) a subcritical rotor design, (2) a sufficient stability margin, and (3) a high integrity rotor balance. Meeting these provisions has required optimization of the mechanical design of the rotor, bearings, rotor supports, damper seals, and housing for successful rotordynamic characteristics.

The initial phase of the DSEFT design has moved the fundamental rotor bending mode (with high rotor strain energy) to 97 percent above the design speed. Consequently, the synchronous resonant response due to the rotor imbalance is almost completely eliminated. The two modes occurring at 96 and 267 percent of the operating speed are low rotor strain energy rigid body pitch and bounce modes of the rotor, as shown in Figures 4.2.1.3-5 and -6, respectively.

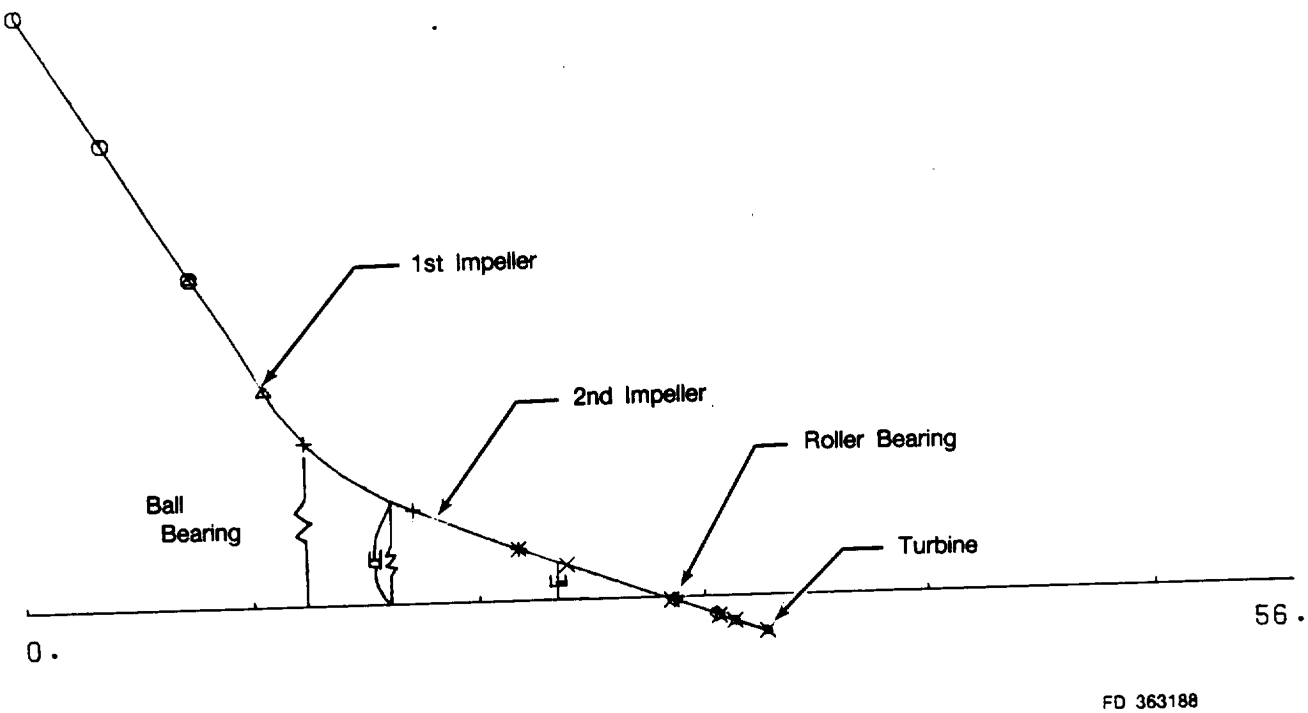

Figure 4.2.1.3-5. STBE Derivative Split Expander Fuel Turbopump Critical Speeds Analysis Showing the Pump Pitch Mode of the Rotor at 96% Design Speed (RPM = 10815)

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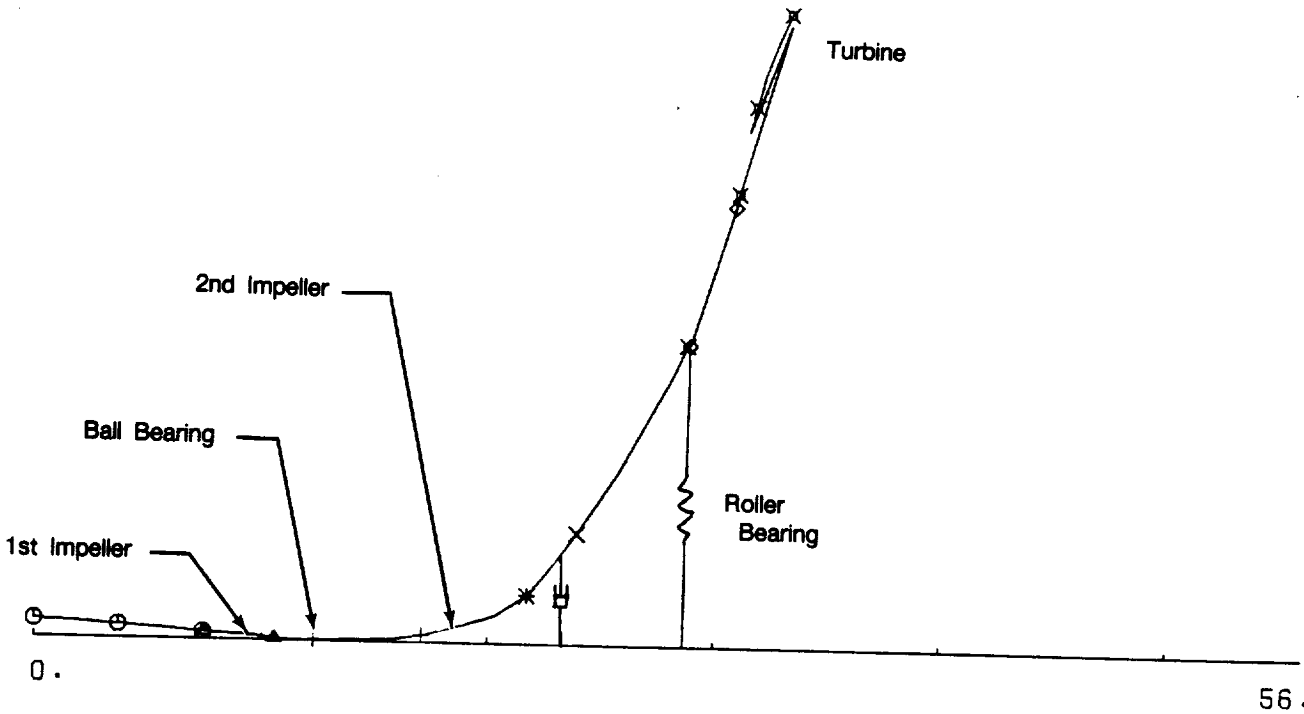

Figure 4.2.1.3-6. STBE Derivative Split Expander Fuel Turbopump Critical Speeds Analysis Showing the Turbine Bounce Mode of the Rotor at 267% Design Speed (RPM = 30122)

Rotor bearing stiffness plays a very important part in the dynamic behavior of all turbomachinery. In high-pressure rocket turbopump designs, P&W realizes the need for the combined rotor support system stiffness (bearing, carrier, and backup structure) to approach or exceed the relative stiffness of the rotor structure to minimize rotor strain energy.

The rotor critical speed analysis has been used to set initial design requirements for each bearing stiffness. The pump and bearing is a large diameter, high load capacity ball bearing with minimum IRC and deadband. The turbine end bearing is a large diameter, high load capacity roller bearing with negative IRC. Without exceeding successful P&W bearing DN experience, high rotor support stiffness coupled with a stiff rotor design in this application, ensure that successful rotordynamic design criteria are met.

Rotordynamic stability is further improved by eliminating the subsynchronous rotor excitation associated with the first rotor mode below 50 percent of the design speed. This has been accomplished by moving the first rotor rigid mode to 96 percent of the design speed and by the use of roughened stator damper seals at the impeller interstage locations.

Rotordynamic stability analysis will be used as a design tool to determine the final damper seal configuration requirements for optimized system dynamics. However, the DSEFT is designed such that damper seals are not critical to the dynamics of the rotor system. Roughened stator damper seals are included in the DSEFT design. Each of the seals is designed for high damping, moderate stiffness, and minimal leakage. The incorporation of damper seals in the turbopump provides reduced synchronous response throughout the operating range resulting in low dynamic bearing loads and rotor deflections, sufficient margin on the OSI, and additional

rotor load support. Parametric studies on the interstage damper seal locations will be presented in design Phase B.

Balance provisions and techniques used for DSEFT are identical to those used for the DSEOT.

A critical speed summary for DSEFT is provided below.

W_{cr} *(rpm)*	*% Design Speed*	*% Rotor Strain Energy*	*Mode Description*
10815	96.0	24.5	Pump Pitch
16704	148.0	88.8	1st Bending
30122	267.0	32.3	Turbine Bounce

4.2.1.4 Combustor

The derivative STBE split expander engine minimum chamber volume, injector design, and acoustic liner design were determined using the procedures outlined in section 4.1.1.4 for the derived STBE engine gas generator cycle. The derivative STBE (split expander cycle) chamber and injector element designs are summarized in Table 4.2.1.4-1. An L* of 41 inches is required to meet the specified 98 percent characteristic velocity efficiency. This is greater than that of the gas generator cycle engines mainly as a result of poorer atomization in the split expander due to less available pressure drop across the injector elements.

Table 4.2.1.4-1. *Derivative STBE Split Expander Combustor and Injector Design*

Chamber L* (Min)-in.	41
Fuel Flow-lb/sec	479.1
ΔP Fuel-psi	62.3
LO_2 Flow-lb/sec	1676.7
ΔP LO_2 psi	43.7
No. of Elements	2360
Spud ID-in.	0.238
Annular Gap-in.	0.0275

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The acoustic liner design of the derivative STBE chamber is given in Table 4.2.1.4-2. The acoustic absorption of the liner is 21.0 percent at the first tangential frequency (773 Hz). This meets the minimum absorption requirements to stabilize combustion in the liner.

4.2.1.4.1 Main Injector

The main injector design uses 2360 coaxial, tangential entry injection elements arranged in a concentric pattern in a 36.960-inch diameter injector face. This type of injector element has consistently demonstrated efficient, stable combustion in all of the P&W high-pressure combustion programs.

Table 4.2.1.4-2. *Derivative STBE Split Expander Acoustic Liner Design*

Chamber Pressure-psi	733.7
Aperture — Gas Temperature-°R	2000
Aperture — Gas Molecular Wt.	21.7
Hole Diameter-in.	0.1
Hole Length-in.	0.35
Area Ratio	0.05
Backing Cavity Depth-in.	0.60
Liner Length-in.	7.24

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The oxidizer injection element is a tube which is closed at one end and has a 0.238-inch ID and a 0.020-inch wall thickness. The oxidizer is introduced into the tube through three slots that are oriented on a tangent to the tube ID. The tangential entry produces a hollow cone spray of liquid oxygen which results in extremely fine atomization, and rapid, stable combustion.

Fuel is introduced through an annulus surrounding each LO_2 injection element. The annulus is formed by the fuel sleeve which is cast integral with the injection element and brazed to the porous faceplate. Fuel enters the injector from the combustion chamber coolant interface, and flows radially inward in the injector manifold which is formed by the interpropellant divider plate and the porous faceplate. At each LO_2 injection element, the fuel is directed into the individual fuel annuli by four radial slots in the fuel sleeve. The fuel is then discharged from a 0.0275 in.^{2} annulus surrounding each LO_2 injection element. The faceplate is fabricated from a porous material, woven wire product consisting of Haynes 230 cobalt alloy, allowing approximately five percent of the fuel which is introduced into the injector to flow through the injector face to achieve faceplate durability.

The main injector assembly is fabricated from fine grained cast and HIP Inconel 718 with cast injection elements integral with the propellant divider plate. The injector design provides for a center-mounted torch igniter and also is configured to contain the engine gimbal thrust structure.

The main injector assembly is shown with its key dimensions in Figure 4.2.1.4-1. The main injector element and element pattern are shown in Figures 4.2.1.4-2 and -3, respectively.

4.2.1.4.2 Combustion Chamber

The combustion chamber is regeneratively cooled by fuel from the high-pressure pump discharge. The fuel enters the tubular cooling jacket through the inlet manifold below the throat. The coolant then flows forward, counter to the gas path flow, to the throat. The fuel then cools the chamber wall, is collected and exits through the toroidal shaped manifold. This flow configuration provides the coolest fuel at the throat where wall heat flux is highest. The combustion chamber is shown in Figure 4.2.1.4-4.

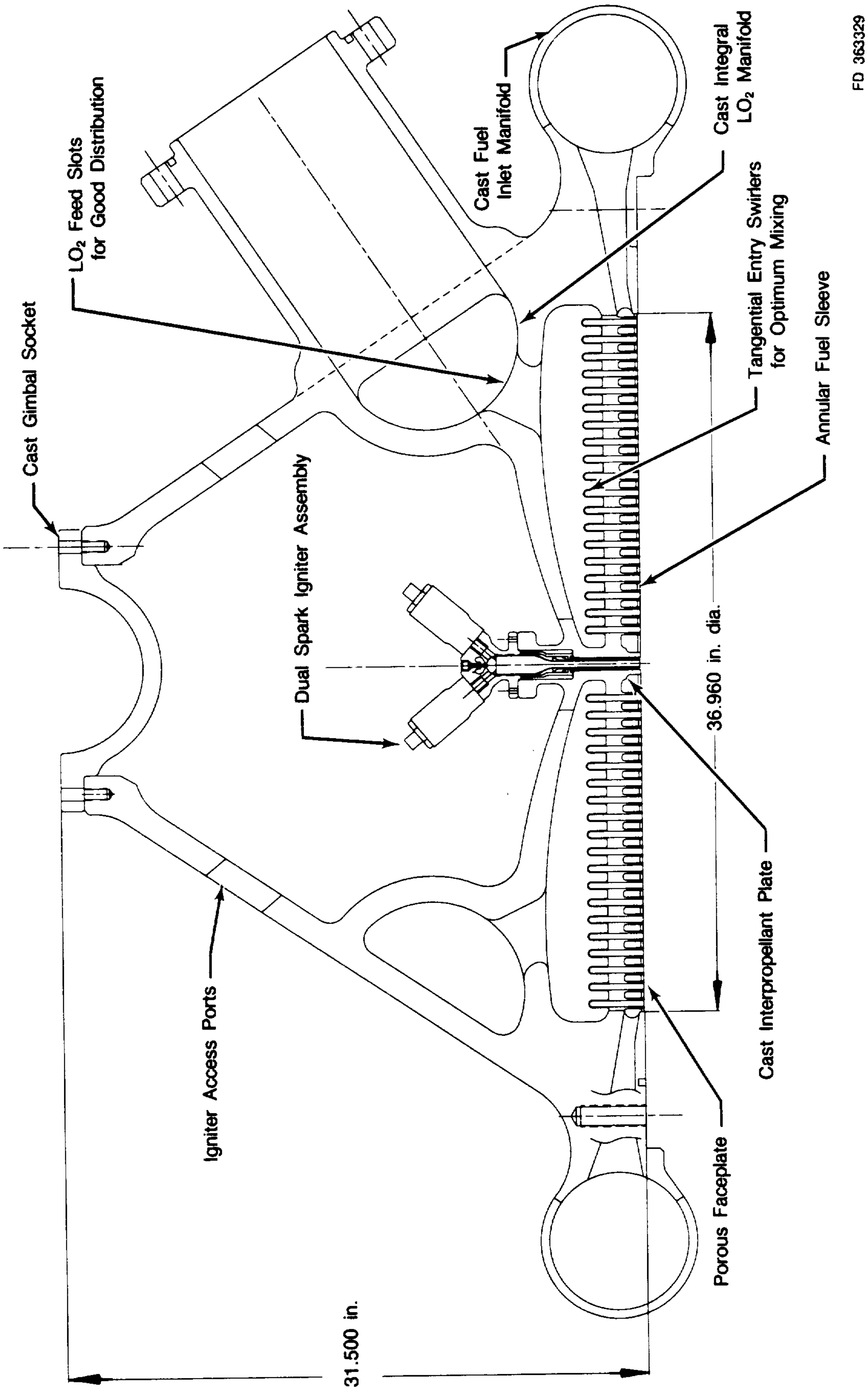


Figure 4.2.1.4-1. STBE Derivative Split Expander Main Injector

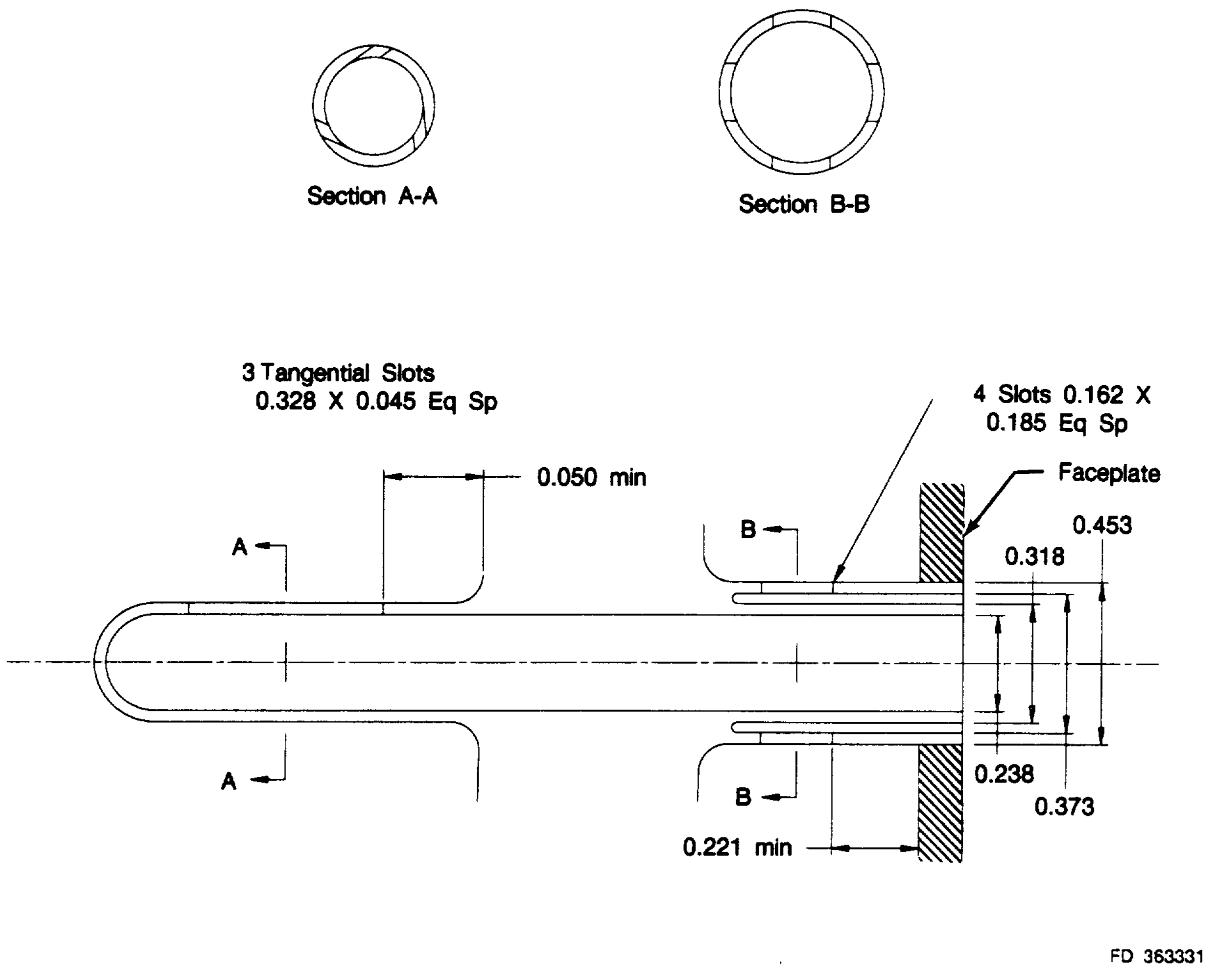


Figure 4.2.1.4-2. STBE Derivative Split Expander Main Injector Element

The thrust chamber design uses a brazed assembly of 720 double tapered, constant wall thickness Haynes 230 tubes. The chamber extends to a nozzle expansion area ratio of 2.40:1, has an injector diameter of 36.96 inches and a throat diameter of 26.14 inches, with a corresponding contraction ratio of 2.0. Acoustic apertures are located within the braze joints at the forward end of the chamber. A counterflow cooling system that uses 53 percent of the methane fuel flow is used to regeneratively cool the nozzle. The coolant tube dimensions are sized to meet the heat transfer and cycle requirements at the 600K lbf sea level thrust at 734 psia chamber pressure design point and reflect the following design guidelines.

- Maximum stress < 0.2 percent yield strength.
- Ultimate tube temperature margin > 375R.
- Coolant Mach number < 0.5.
- Cooling enhancement from tube curvature.

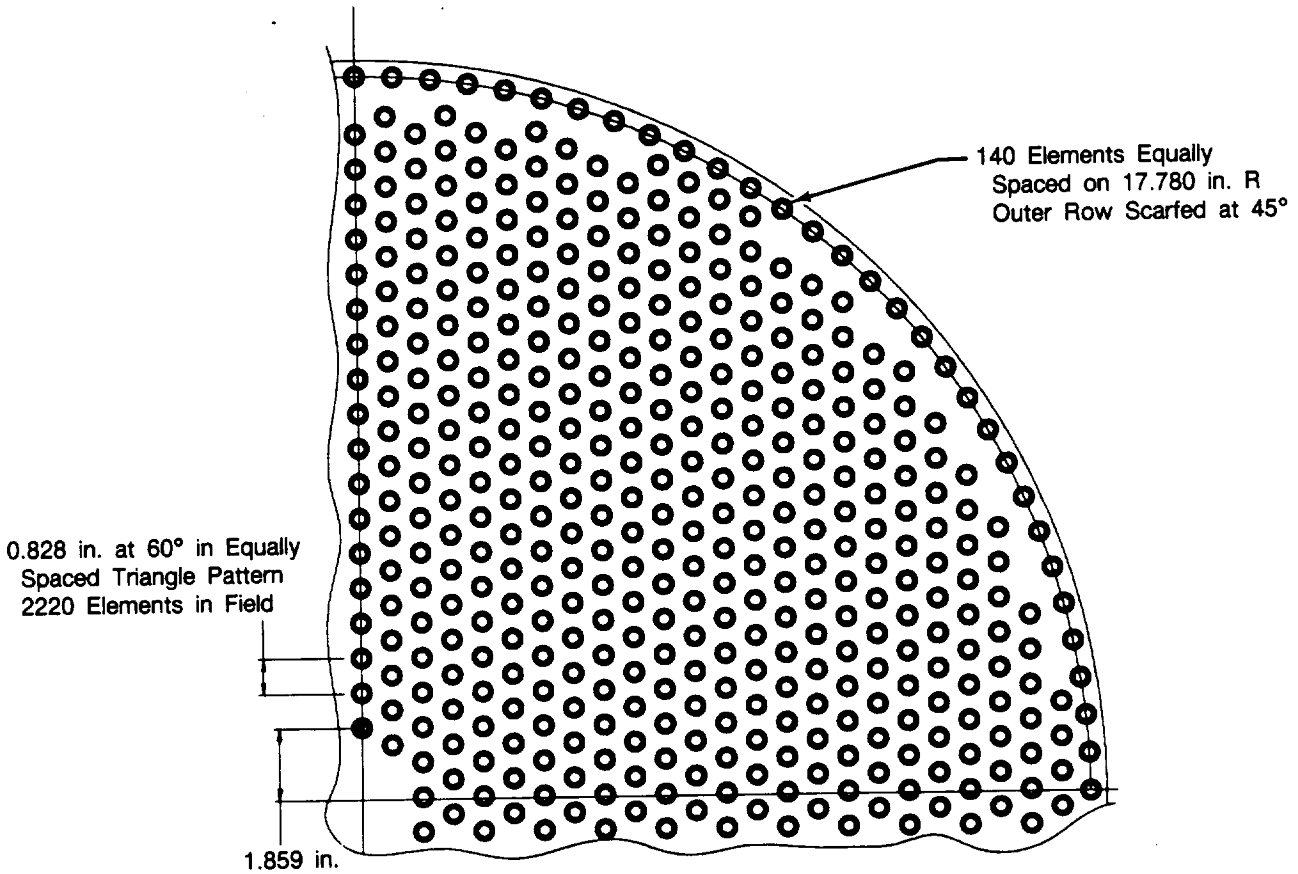


Figure 4.2.1.4-3. STBE Derivative Split Expander Main Injector Pattern

Figure 4.2.1.4-5 summarizes the throat chamber contour and tube geometry.

The methane coolant enters the tube assembly at 251 R and 5450 psia and exits at 572 R and 4572 psia. The maximum predicted values of hot wall temperature and heat flux are 2120 R and 20 Btu/in.^{2}-sec, respectively. The highest calculated coolant Mach number is 0.13. Figure 4.2.1.4-6 summarizes the predicted thermal performance characteristics for the thrust chamber.

4.2.1.4.3 *Regeneratively Cooled Nozzle*

The regeneratively cooled nozzle is constructed from 1180 Haynes 230 nickel alloy tubes, brazed together and surrounded by a structural shell of closed cell elastomeric foam with a filament wound composite overwrap. This shell is also designed to carry all hoop loads. Coolant inlet and exit manifolds, fabricated from cast Haynes 230, are brazed to each end of the chamber, thereby forming the entire nozzle assembly.

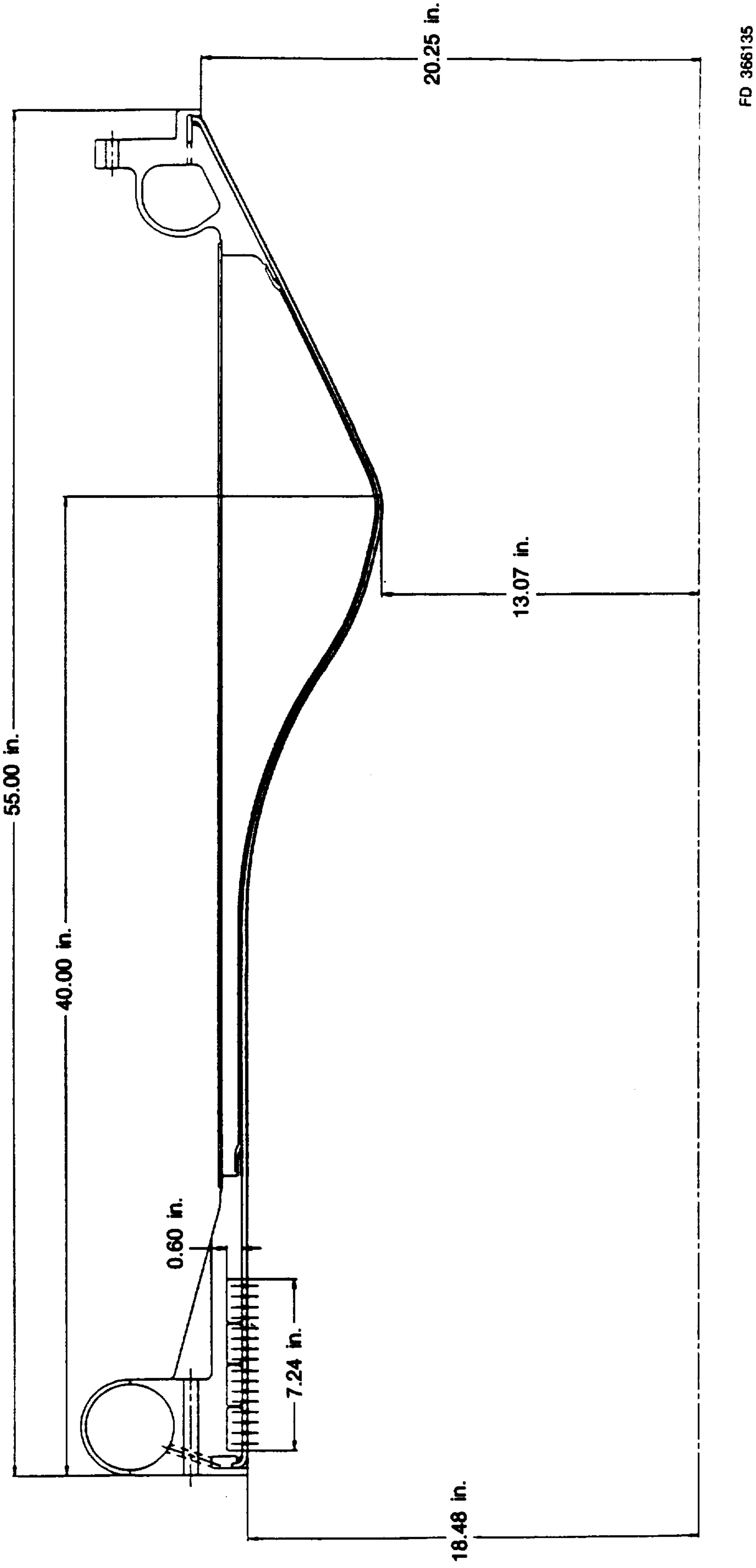
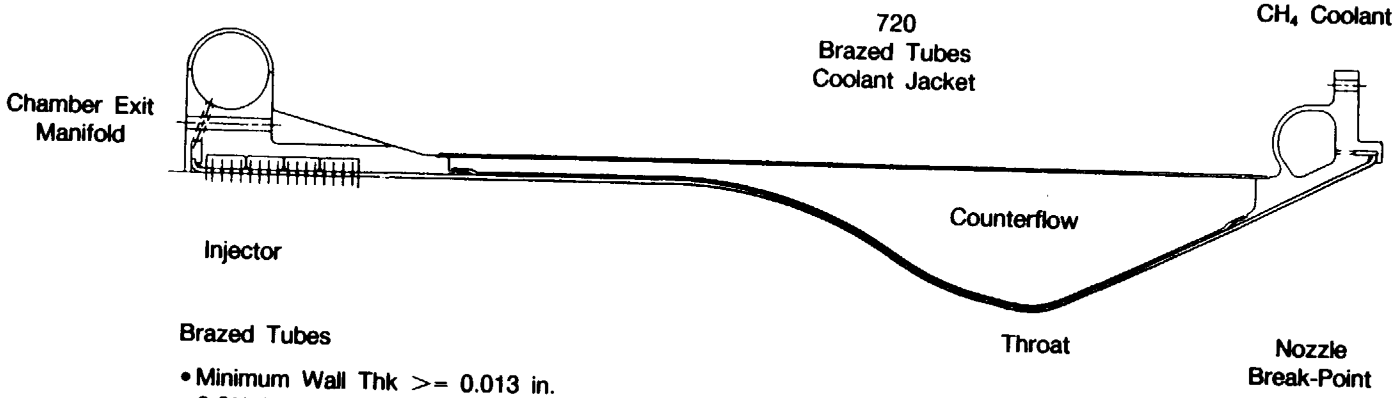


Figure 4.2.1.4-4. STBE Derivative Split Expander Combustion Chamber



Brazed Tubes

- Minimum Wall Thk >= 0.013 in.
- 0.2% Yield Strength/Tube Stress >= 1.0
- Coolant Mach No. <= 0.5
- Maximum Wall Temperature Limit <= 2260 R
- Minimum UTTM >= 375 R

Thrust Chamber Contour Data

- Chamber Length = 28 inches
- Divergent Nozzle Length = 15 inches
- Throat Diameter = 26.14 inches
- Injector Diameter = 36.96 inches
- Contraction Ratio = 2.0
- Divergent Nozzle Area Ratio = 2.40
- L_* = 47.5 inches
- η_{C*} (Throat) = 0.99
- Number of Tubes = 720
- Liner Construction - Brazed Tubes
- Liner Material - Haynes 230

Thrust Chamber Cooling Tube Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)	Aspect Ratio
15.00	20.25	0.175	0.194	0.013	1.109
9.94	17.72	0.153	0.169	0.013	1.102
4.88	15.29	0.132	0.148	0.013	1.121
1.85	13.91	0.120	0.155	0.013	1.290
0.00	13.07	0.113	0.159	0.013	1.413
-2.20	13.25	0.114	0.158	0.013	1.385
-4.22	13.77	0.119	0.156	0.013	1.312
-7.26	15.27	0.132	0.148	0.013	1.124
-10.80	17.16	0.148	0.163	0.013	1.101
-17.38	18.48	0.160	0.176	0.013	1.099
-28.00	18.48	0.160	0.177	0.013	1.104

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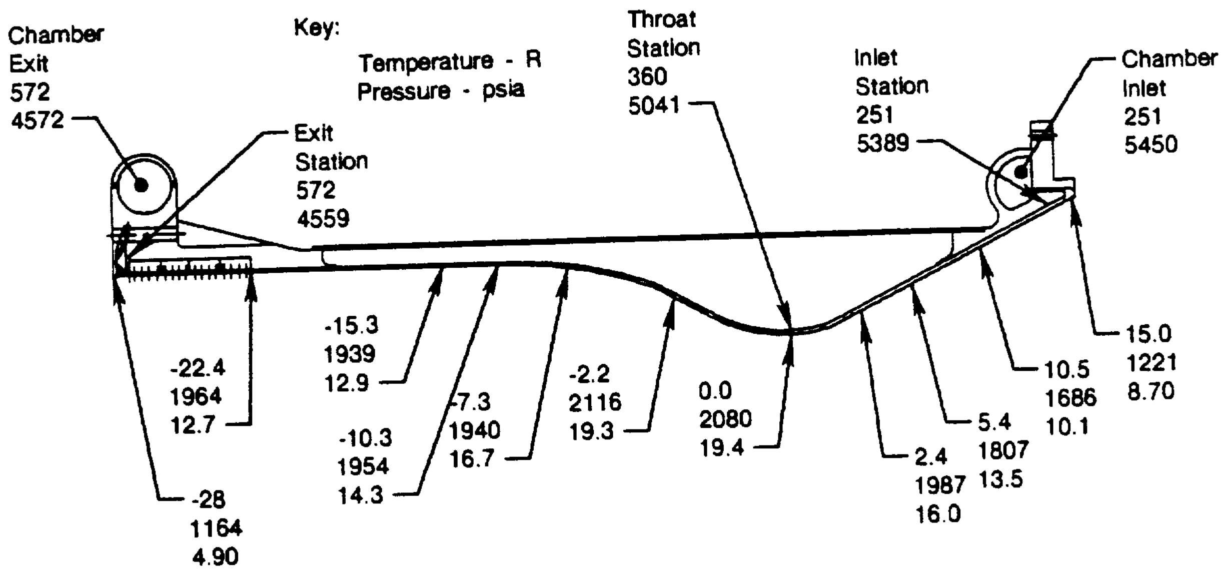
Figure 4.2.1.4-5. STBE Derivative Split Expander Thrust Chamber Cooling Design Configuration



Coolant Performance

Thrust = 100%

M_{cool} = 276 lbm/sec

Hot Wall Temperature and Heat Flux

Key:

Axial Location - in.
Wall Temp - R
Heat Flux - Btu/in.^{2}-sec

Thrust Chamber Heat Transfer Performance

SL Thrust - lbf	600K
Chamber Pressure - psia	734
Propellant	LO_2/CH_4
O/F Ratio	3.50
Coolant Flow - lbm/sec	276
Inlet Temperature - R	251
Exit Temperature - R	572
Coolant Heat Pickup - Btu/sec	72,348
Inlet Pressure - psia	5450
Exit Pressure - psia	4572
Pressure Drop - psid	878

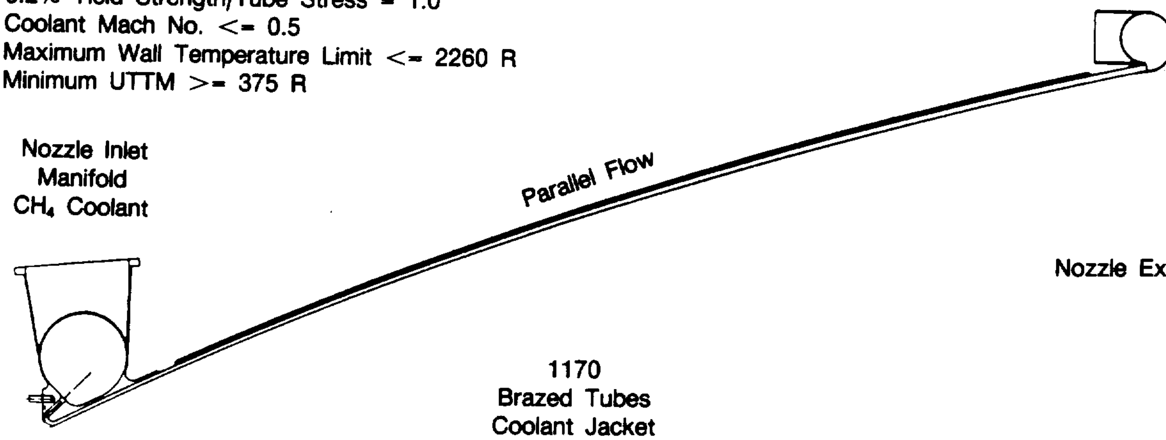
FDA 363812

Figure 4.2.1.4-6. STBE Derivative Split Expander Thrust Chamber Heat Transfer Performance Summary

The regeneratively cooled nozzle design uses a brazed assembly of 1170 single tapered, constant wall thickness Haynes 230 tubes, is 80-inches long and extends from an expansion area ratio of 2.40:1 to an exit area ratio of 13.5:1. A parallel flow cooling system that uses 53 percent of the methane fuel flow is used. The methane is used to cool the thrust chamber prior to cooling the nozzle. The nozzle tube dimensions are sized to meet the heat transfer and cycle requirements at the 600K lbf sea level thrust at 734 psia chamber pressure design point. The tube geometry reflects the following design guidelines.

- Maximum stress < 0.2 percent yield strength.
- Ultimate tube temperature margin > 375R.
- Coolant Mach number < 0.5.

Figure 4.2.1.4-7 summarizes the nozzle contour and tube geometry.



Nozzle Contour Data

- Nozzle Length = 80 in.
- Inlet Expansion Ratio = 2.40
- Exit Expansion Ratio = 13.5
- Number of Tubes = 1170
- Nozzle Construction - Brazed Tubes
- Nozzle Material - Haynes 230

Nozzle Cooling Tube Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)	Aspect Ratio
15.00	20.25	0.107	0.180	0.013	1.680
25.11	25.15	0.133	0.165	0.013	1.235
34.21	29.15	0.155	0.171	0.013	1.103
46.35	33.86	0.180	0.199	0.013	1.105
55.45	37.07	0.198	0.219	0.013	1.108
64.55	39.95	0.213	0.236	0.013	1.109
76.69	43.45	0.232	0.257	0.013	1.110
85.79	45.79	0.245	0.272	0.013	1.111
95.40	48.01	0.257	0.286	0.013	1.114

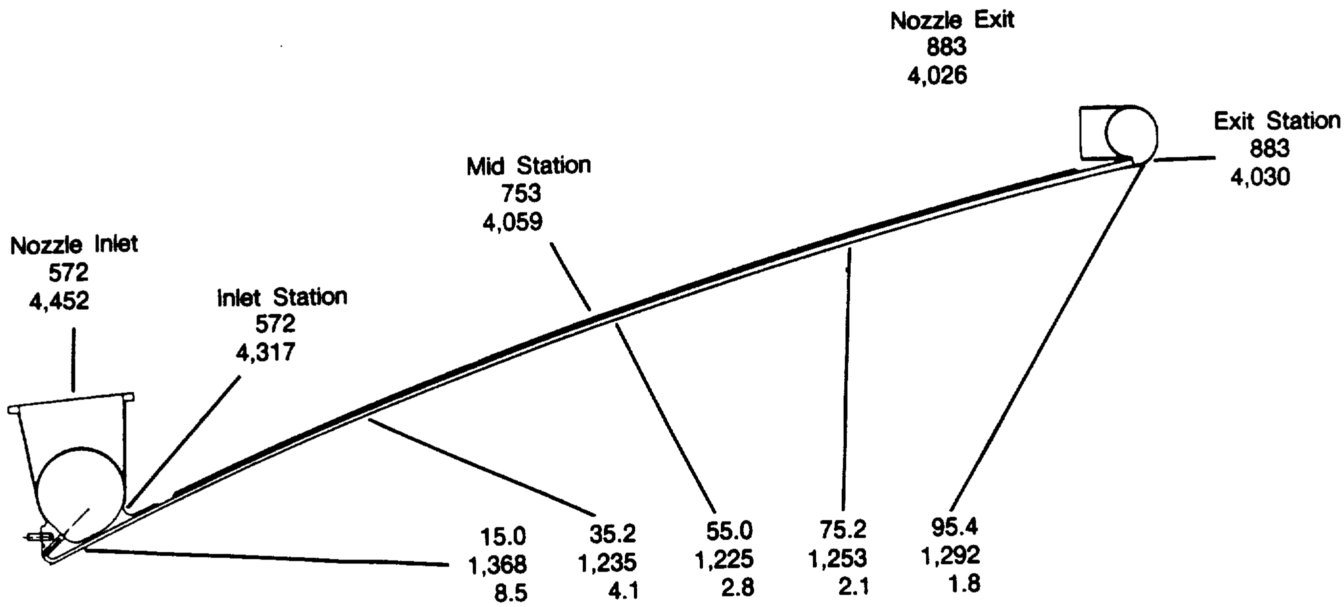
FD 363813

Figure 4.2.1.4-7. STBE Derivative Split Expander Nozzle Cooling Design Configuration

The methane coolant enters the tube assembly at 572 R and 4452 psia and exits at 883 R and 4030 psia. The maximum predicted values of hot wall temperature and heat flux are 1370 R and 8.5 Btu/in.^{2}-sec, respectively. The highest calculated coolant Mach number is 0.14. Figure 4.2.1.4-8 summarizes the predicted thermal performance characteristics for the regeneratively cooled nozzle.

4.2.1.4.4 Torch Igniter

A continuous burning torch igniter was chosen for use in the main combustion system because of the simplicity of the design and reliability in tests. The igniter configuration employed evolved from development efforts since 1957 at Pratt & Whitney and is based on experience gained from the successful RL10 and XLR-129 engine programs.



Coolant Performance
Thrust = 100%
M_{cool} = 276 lbm/sec

Nozzle Heat Transfer Performance

SL Thrust - lbf	600K
Chamber Pressure - psia	734
Propellant	LO_2/CH_4
O/F Ratio	3.50
Coolant Flow - lbm/sec	276
Inlet Temperature - R	572
Exit Temperature - R	883
Coolant Heat Pickup - Btu/sec	67,006
Inlet Pressure - psia	4,452
Exit Pressure - psia	4,026
Pressure Drop - psid	426

Hot Wall Temperature and Heat Flux

Key:
Axial Location - in.
Wall Temp - R
Heat Flux - Btu/in.^{2}-sec
Temperature - R
Pressure - psia

FD 363814

Figure 4.2.1.4-8. STBE Derivative Split Expander Nozzle Heat Transfer Performance Summary

In the main combustion chamber, the torch is mounted axially in the center of the injector, directing the torch down along the centerline of the combustion chamber.

The construction of the torch assembly is discussed in Space Transportation Main Engine Configuration Study, P&W FR-19830-1 Volume II, page 93.

4.2.1.5 Controls

The STBE controls consist of sensors, interconnects, a controller, actuators, propellant valves, ancillary valves, and a health monitor. The functional layout of the STME controls components is shown on Figure 4.2.1.5-1. The controller time sequences the valves for engine control and maintains engine safety by sensing hazards and taking corrective action. A single

electromechanical actuator drives the turbine bypass valve. The jacket bypass, main oxidizer, and nozzle shutoff valves are helium actuated. The turbine bypass valve is a sleeve type valve, while the main oxidizer, jacket bypass, and nozzle shutoff valves use similar poppett type valves. The health monitor is integrated with the controller but electrically isolated to prevent health monitor faults from propagating into the controller and jeopardizing engine safety.

Engine thrust is regulated by trimming the turbine bypass valve while engine mixture ratio is regulated by trimming the main oxidizer valve. Oxidizer flow shutoff is provided by the main oxidizer valve while positive fuel flow shutoff is provided by the nozzle shutoff valve and jacket bypass valve.

Requirements used to establish a control and monitoring system concept are shown in Table 4.2.1.5-1.

4.2.1.5.1 Control/Health Monitor Conceptual Architecture

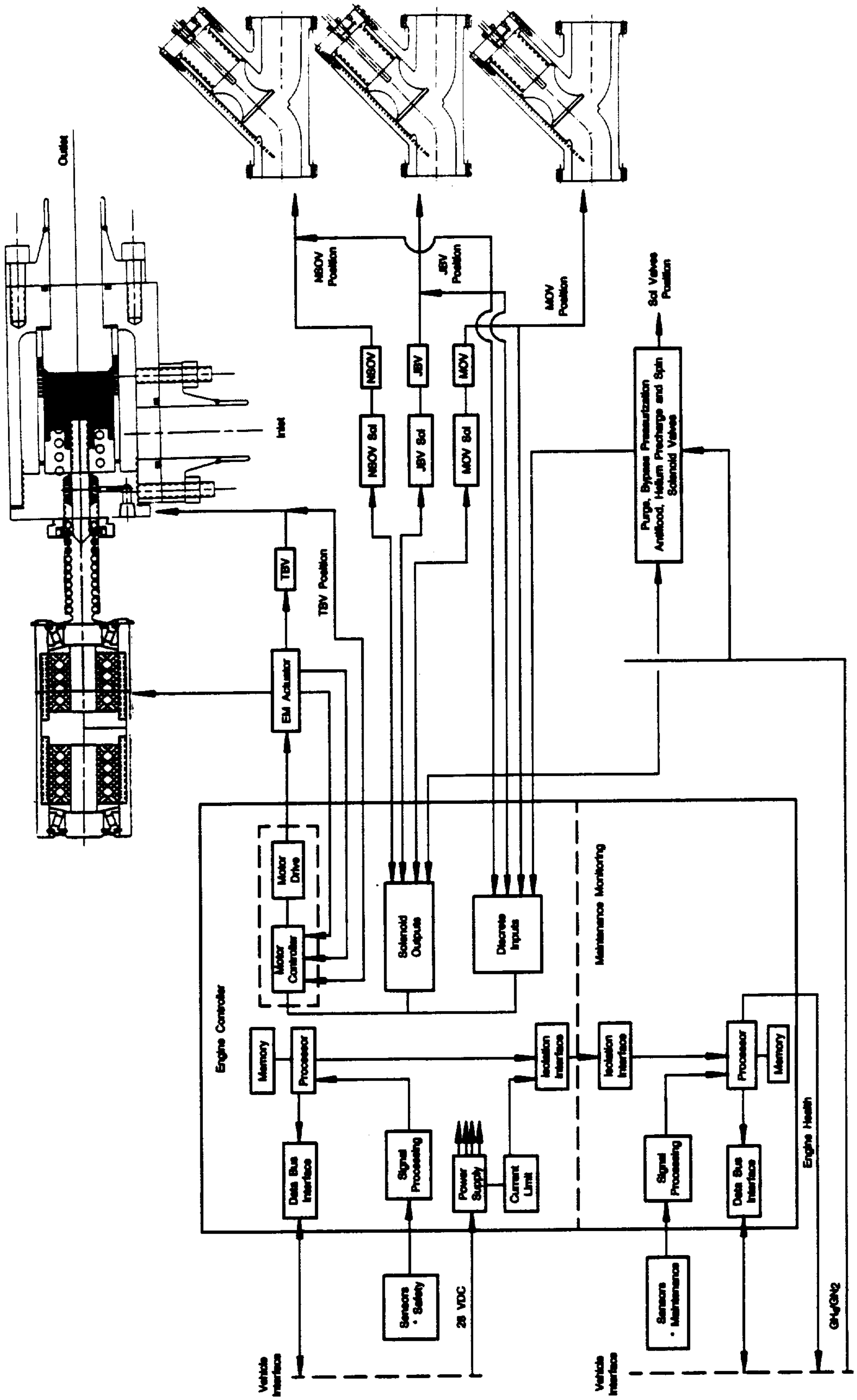
Conceptually the controller/health monitor is comprised of two functions: (1) control and safety monitoring and 2) maintenance monitoring. Control functions are those required to start, maintain normal operating conditions and shutdown the engine. Safety monitoring consists of real time engine evaluation to determine if an emergency shutdown is required. Maintenance monitoring looks at functional and physical characteristics which include many that are not flight critical, but real time definition is necessary to properly schedule maintenance.

The STBE engine uses a simplex, full authority digital electronic engine control with dual channel input/output (I/O). A single channel control with an effector system designed to direct engine shutdown upon loss of controller function meets the fail safe design requirement. Controller reliability requirements are met with dual I/O interfaces which receive inputs from dual sensors with the information being processed by a single microprocessor.

The output interface supports dual-wound solenoids and a dual channel electromechanical actuator interface. One of the two solenoid windings in each device has the capacity for solenoid operation in the event that one winding fails opens. Shorted solenoid switches are accommodated by switching both high and low sides of the solenoid. The electromechanical actuator (EMA) interface is a dual active effector system with single processor control. Under normal conditions, each output interface provides one half the drive signal necessary for actuator control. If one of the EMA interfaces becomes inoperative, the current drivers in the inoperative interface are depowered and the gain in the remaining interface is doubled to provide full control capability. This dual active interface provides smooth transfers from dual channel operation to single channel operation.

Actuator loop failure detection is provided by current wraparound, feedback failure detection and open-loop detection. Current wraparound is provided by measuring actuator winding current and comparing the result to the requested value.

Feedback failures occur if the actuator position sensors produce an erroneous result to the controller. Feedback failure detection is provided by detecting out-of-range readings or detecting a difference between the dual sensor readings. Open-loop detection is provided by comparing the requested actuator position to the measured position. The error between the request and feedback is measured over a period of time and compared to a threshold value. If the measured actuator error is above the threshold value, an open-loop failure is declared. In the event that an actuator malfunction cannot be isolated to a given interface, an engine shutdown is effected by the system logic.



FD 366100

Figure 4.2.1.5-1. Engine Control and Health Monitor System Functional Concept Meets All Requirements With Low-Cost Approach

Table 4.2.1.5-1. Control System Requirements

Requirement	Engine Requirement	Control System Requirement
Thrust — lb	706500	±3.0%
Mixture Ratio NPL	3.5	±3.0%
Failure Accommodation	Fail Safe	Redundancy Management
Start	Self Contained Control Accel to RPL: 5 sec	Control Response
Shutdown	Decel to Zero Thrust in TBD sec	Control Response

R19830/47

An initiated built-in-test (IBIT) mode is provided by the controller to detect faults during prestart launch pad operations. In the IBIT mode, the controller is able to sequence solenoid valves and electromechanical actuators throughout their operating range. This feature enhances mission reliability by providing a low cost method for testing the system prior to launch.

The health monitoring system works as an interface between the electronic control, engine sensors, and the vehicle avionics while transmitting real time data to the Vehicle Health Monitoring System (VHMS). Safety monitoring is performed by the electronic control with any performance or anomaly information passed to the maintenance monitoring unit through an isolation interface. Instrumentation not critical to flight operation is processed by maintenance monitoring electronics. Maintenance monitoring information is transmitted to the vehicle independently of the control.

4.2.1.5.2 Controller Hardware Approach

Highlights of the control/health monitoring system architecture include modular design of the engine control functional requirements. The system level design includes control of discrete inputs and outputs (solenoids and switches), actuator positioning, sensor signal processing and control law processing. This system design is implemented using state of the art hardware which provides a low risk, low cost flexible control.

Current plans are to provide a control design that meets reliability requirements with Class B components. By using these MIL-STD components and proper redundancy management, the reliability requirements can be achieved without the cost penalty of Class S components. With the advent of microelectronics, multiple channel controls are viable options without paying a significant weight penalty. Multiple channel controls will be considered during Phase B as a way to improve life cycle cost.

4.2.1.5.3 Vehicle Interface Definition

Independent vehicle interfaces are supported by both the engine control and health monitor. Independence is necessary to ensure faults in the maintenance data bus from causing a fault in the control data bus. These data buses will be designed to be compatible with the vehicle data bus selection. The only identified differences will be those that address flight criticality. The engine controller interface will be updated to meet different flight safety requirements.

Isolated interfaces between control and maintenance monitors were selected to support the integrated design concept. The key to these interfaces is to incorporate failure containment regions. Failure containment is accomplished through design.

4.2.1.5.4 Actuators/Valves

An extensive trade study was conducted to select valve and actuator types based upon an assessment of cost, reliability, performance and hardware commonality. Low cost was ranked as the primary selection criteria with manufacturability, design simplicity and maintainability all being considered cost drivers. The study considered pneumatic, hydraulic and electromechanical actuators as well as sleeve, poppett, ball, and butterfly valves. From this study, the following configurations were selected.

4.2.1.5.4.1 *Main Oxidizer Valve (MOV)*

The MOV is an on/off valve that is located downstream of the oxidizer pump and upstream of the thrust chamber in the oxidizer line. Its function is to control liquid oxidizer flow to the thrust chamber and thereby control the engine oxidizer/fuel mixture ratio. To meet the engine start and throttling requirements, the valve requires only one full open and one full closed position. The valve must provide $\pm 10\%$ trimmability at the open position for engine mixture ratio trimming during engine acceptance testing. A poppett valve has been selected as the lowest cost valve type which will meet all requirements. Also, as stated in the gas generator valve section, the poppett lends itself to precision trimming at the 90 percent open position, allowing accurate mixture ratio trimming. Since the valve has only two operating positions, full open and full closed, a translating helium piston actuator has been selected as the lowest cost option meeting all requirements. The actuator position will be controlled through a solenoid valve which is electrically scheduled by the engine controller. Discrete actuator position switches provide valve position feedback to the controller for preflight checkout as well as for in-flight operation.

MOV Option No. 1 — To further reduce system cost and improve the reliability by removing components from the system, an optional propellant actuated MOV has been identified. The poppett valve may be pressure balanced and spring loaded such that the difference between the oxidizer pump inlet pressure and the pump outlet pressure serves as the actuation force on the MOV. This configuration restricts the MOV from easily being checked out during the preflight inspections; however, it reduces the potential of an uncommanded valve closure during main stage operation by removing the solenoid actuator and replacing it with a force balanced poppett assembly. Thus, the MOV will not close until the oxidizer pump pressure delta falls below 300 psid, eliminating the solenoid and actuator failure mode in which the pump is overpressurized as a result of MOV closure at main stage operation.

MOV Option No. 2 — The MOV may also be electromechanically actuated to provide active mixture ratio trim during engine operation. Using the pressure balance technique, the valve loads may be reduced such that the electromechanical actuator used for the turbine bypass valve (TBV) may also be used for the MOV. The characteristics of this actuator are discussed in the Ganged Gas Generator Valves/Actuation section (4.1.1.6.4.1).

4.2.1.5.4.2 *Nozzle Shutoff Valve (NSOV)*

The NSOV is an on/off valve that is located downstream of the fuel pump second stage discharge and upstream of the nozzle in the fuel system. Its function is to control fuel coolant flow into the nozzle. To provide maximum cost benefit, a poppett type valve similar to the MOV has been selected. While pressure drop and weight could be improved using a ball valve design in this location, these factors have been traded for the simpler, lower cost poppett which also provides similarity with the MOV and the cost benefits which accompany similarity in

development, production, and logistics support. The actuator is identical to that of the MOV providing additional system commonality. The actuator position will be controlled through a solenoid valve which is electrically scheduled by the engine controller. Discrete actuator position switches provide valve position feedback to the controller for preflight checkout as well as for in-flight operation.

4.2.1.5.4.3 Jacket Bypass Valve (JBV)

The JBV is an on/off control valve that is located downstream of the first-stage fuel pump and before the mixer. Its function is to control the fuel flow which bypasses the nozzle coolant jacket. To provide maximum cost benefit, a poppett type valve identical to the NSOV has been selected. While pressure drop and weight could be improved using a ball valve design in this location, these factors have been traded for the simpler, lower cost poppett which also provides commonality with the NSOV and similarity to the MOV and the cost benefits which accompany commonality in development, production, and logistics support. The actuator is identical to that of the MOV providing additional system commonality. The actuator position will be controlled through a solenoid valve which is electrically scheduled by the engine controller. Discrete actuator position switches provide valve position feedback to the controller for preflight checkout as well as for in-flight operation.

JBV Option No. 1 — To further reduce system cost and improve the reliability by removing components from the system, an optional propellant actuated JBV has been identified. The poppett valve may be pressure balanced and spring loaded so that the difference between the oxidizer pump inlet pressure and the pump outlet pressure serves as the actuation force on the JBV. This configuration restricts the JBV from easily being checked out during the preflight inspections, however it reduces the potential of an uncommanded valve closure during main stage operation by removing the solenoid actuator and replacing it with a force balanced poppett assembly. Thus, the JBV will not close until the oxidizer pump pressure delta falls below 300 psid, eliminating the solenoid and actuator failure mode in which the pump is overpressurized as a result of MOV closure at main stage operation.

JBV Option No. 2 — The JBV may also be electromechanically actuated to provide active mixture ratio trim during engine operation. Using the pressure balance technique, the valve loads may be reduced such that the electromechanical actuator used for the turbine bypass valve (TBV), may also be used for the JBV. The characteristics of this actuator are discussed in the Ganged Gas Generator Valves/Actuation section.

4.2.1.5.4.4 Turbine Bypass Valve (TBV)

The TBV is a variable control valve that is located downstream of the nozzle coolant jacket and upstream of the fuel mixer, parallel to the turbine flow path. Its function is to control the amount of fuel flow bypassing the turbine flow path and thereby control engine thrust. As shown in the duty cycle in Figure 4.2.1.5-2 the valve also provides engine shutdown by opening to the maximum area. A right angle inlet to outlet translating sleeve valve was selected as the lowest cost option for this application. To meet the failsafe requirements for benign engine shutdown and to minimize the required actuator force, the TBV is pressure balanced and spring loaded to the open position. Thus, upon loss of actuator input force, the TBV slews to the open position at a rate controlled by the valve force balance. The TBV is spring loaded open for safety in case of an actuator failure.

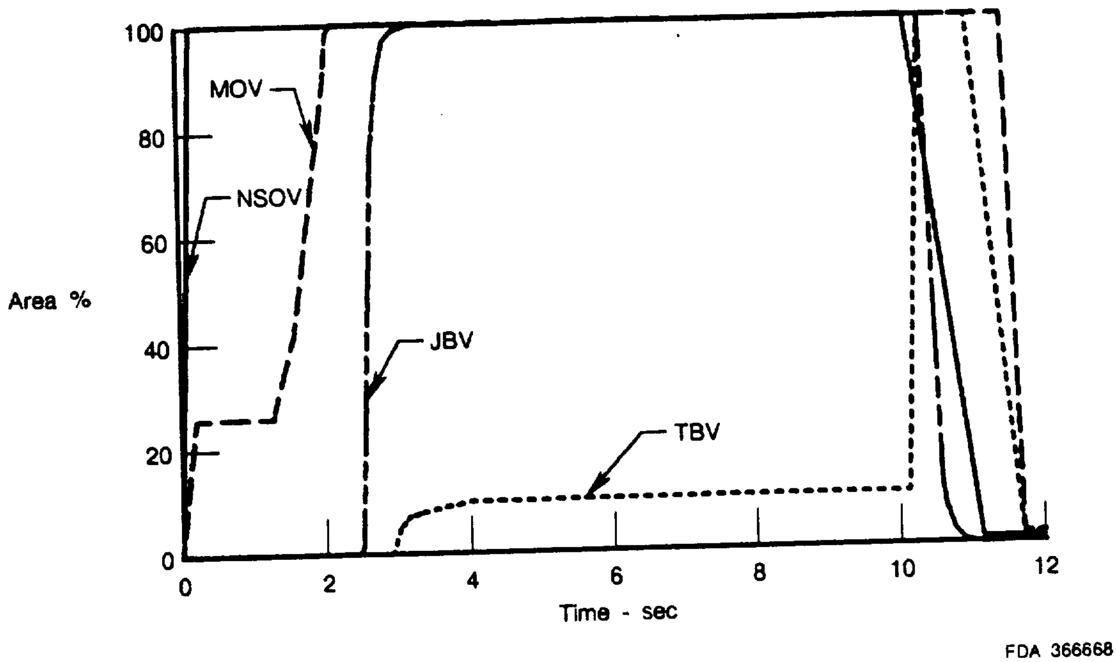


Figure 4.2.1.5-2. STBE Derivative Split Expander Start/Shutdown Valve Sequences

Variable actuation will be achieved by the electromechanical actuator concept discussed in the Ganged Gas Generator Valves/Actuation section (4.1.1.6.4.1).

4.2.1.5.4.5 Ancillary Valves

To provide propellant purging upon engine shutdown, tank pressurization during engine operation, pump interstage dam pressurization and main oxidizer valve bypass, solenoid actuated ancillary valves will be used. In each case the valves are low cost poppett type valves which require only short stroke actuation. For the propellant purge valves, a check valve is located between the poppett and the propellant line to help insure that the propellant is isolated from the helium system. These valves will incorporate commonality when possible, however sizing and failsafe requirements for each valve must be defined before the degree of commonality can be established.

4.2.1.5.5 Operation

4.2.1.5.5.1 Pre-Launch Checkout

All helium and electromechanically actuated engine valves are stroked from full closed to full open to full closed for pre-start valve checkout. The health monitoring system will monitor this pre-launch checkout sequence and identify those line replaceable units (LRU's) which are not within specifications for repair or replacement.

4.2.1.5.5.2 Prestart

(Refer to valve logic schematic in Figure 4.2.1.5-3)

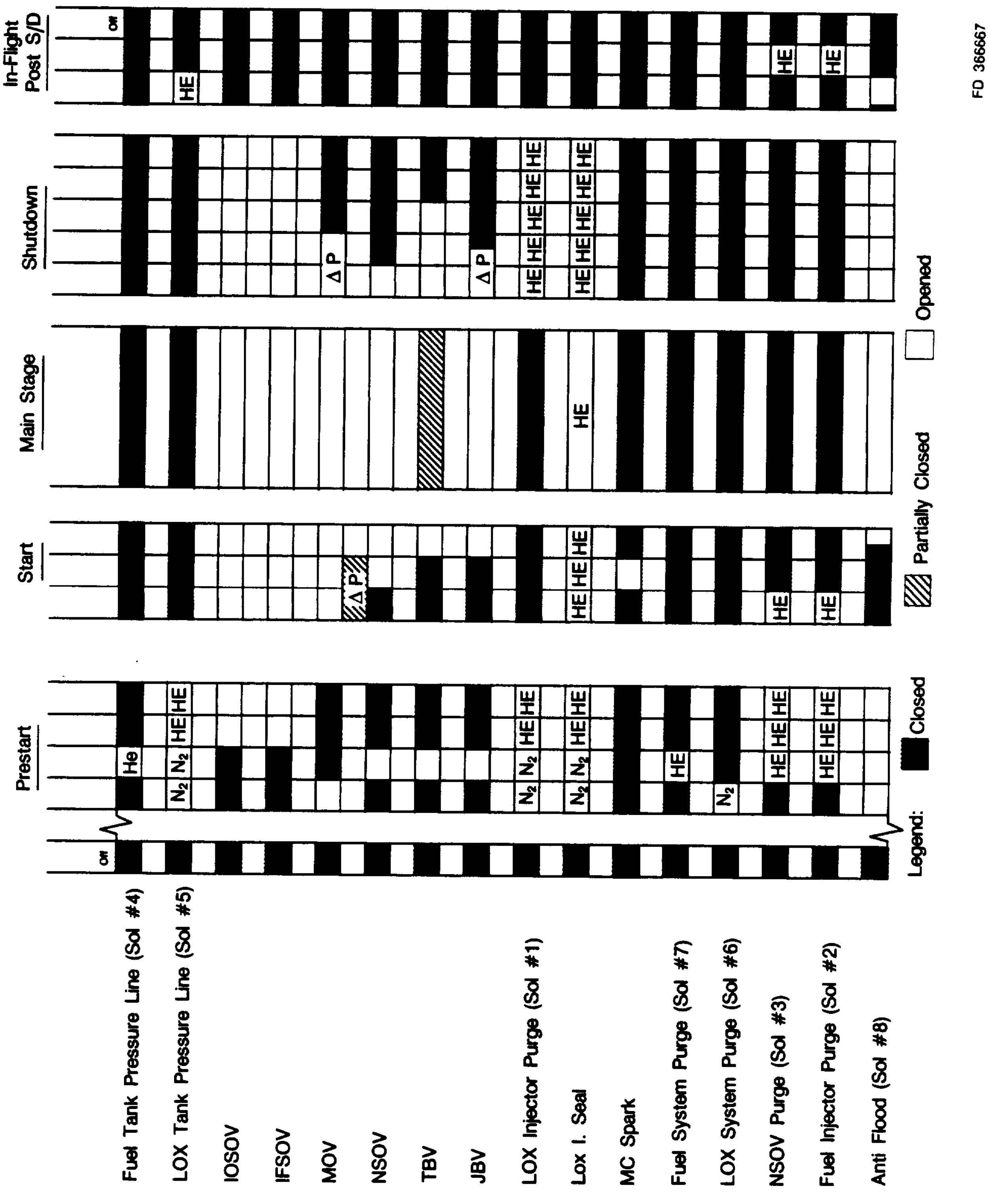


Figure 4.2.1.5-3. STBE Split Expander Valve Cycle Schedule

During prestart, the entire engine fuel and oxidizer systems are purged with GHe and GN_2, respectively. Then following a TBD time interval, the engine control valves are closed. All purging downstream of the fuel side engine valves continues to prevent air (or oxygen) from entering the fuel side volumes when engine start is initiated. As soon as the engine valves are confirmed close, both engine inlet valves are opened to initiate the "cold soak" pump cooldown process. The engine is then held in a "start ready" condition.

4.2.1.5.5.3 Start

(Refer to valve timing in Figure 4.2.1.5-2)

When the engine control receives a start command, the start sequence is initiated. The start sequence begins by opening the MOV to 25 percent area, providing the LO_2 to the LO_2 injector and the igniter. During this time, the fuel purge solenoids Nos. 2 and 3 are open to prevent oxygen from entering the fuel system. After 0.6 second, the NSOV is opened and the igniter spark is turned on. The fuel path from the NSOV to the torch igniter is shorter than the path to the fuel injector. As a result, the torch is lit before fuel reaches the injector. The NOSV opening is timed so that the LO_2 injector is filled prior to fuel entering the main chamber. After filling, the spark igniter is turned off and the fuel purge solenoids are closed.

As fuel exits through the cooling jacket (which is initially at ambient temperature), it is heated to approximately metal temperature, providing energy to accelerate the turbines. As the combustion energy becomes available, acceleration increases and the pump pressures will increase enough to open the MOV and the JBV. The TBV will open as a function of turbine pressure drop as chamber pressure approaches RPL. (See Figures 4.2.1.5-4 through -6)

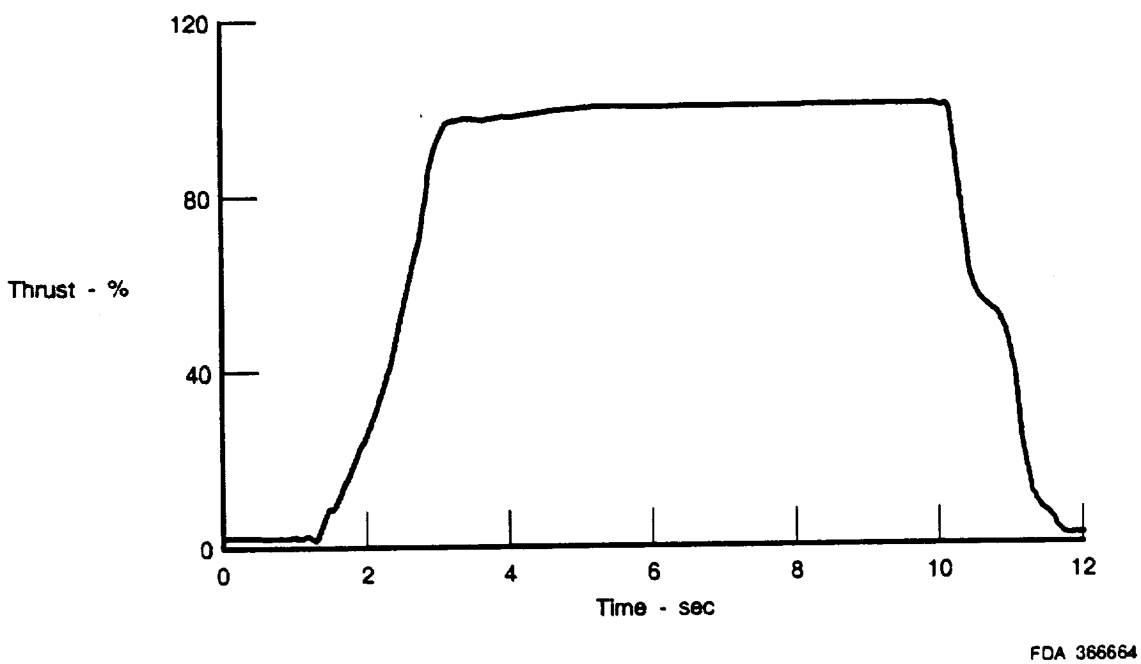


Figure 4.2.1.5-4. STBE Split Expander Start/Shutdown Transient Analysis

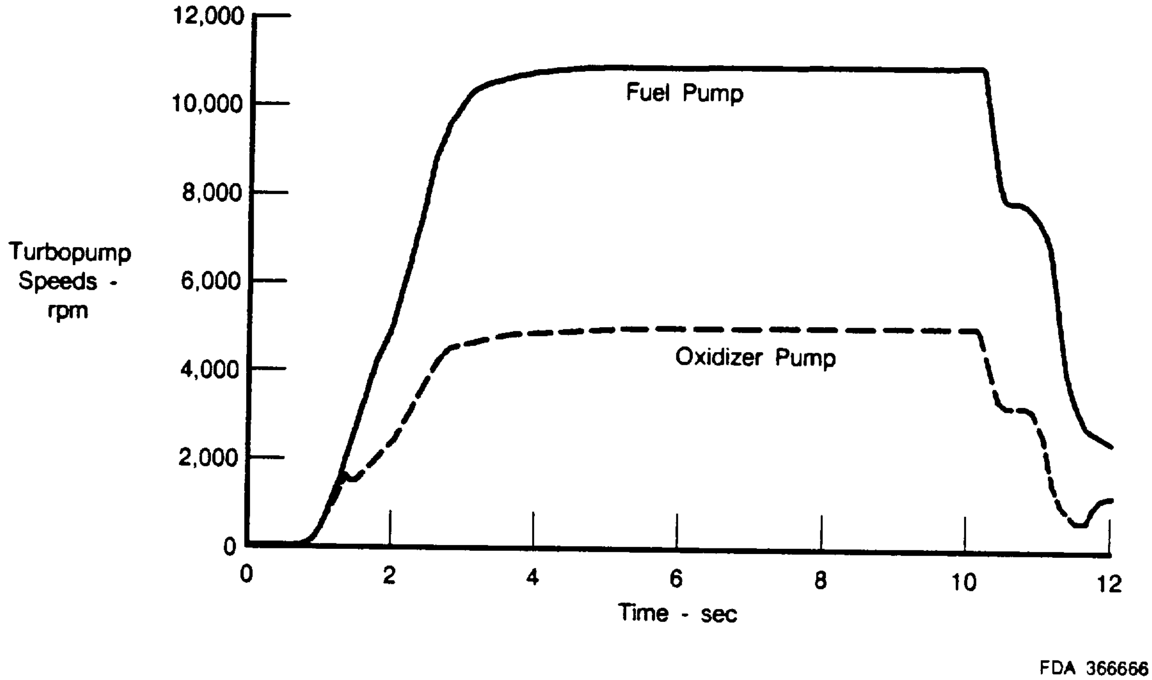


Figure 4.2.1.5-5. STBE Split Expander Turbopump Start/Shutdown Transient

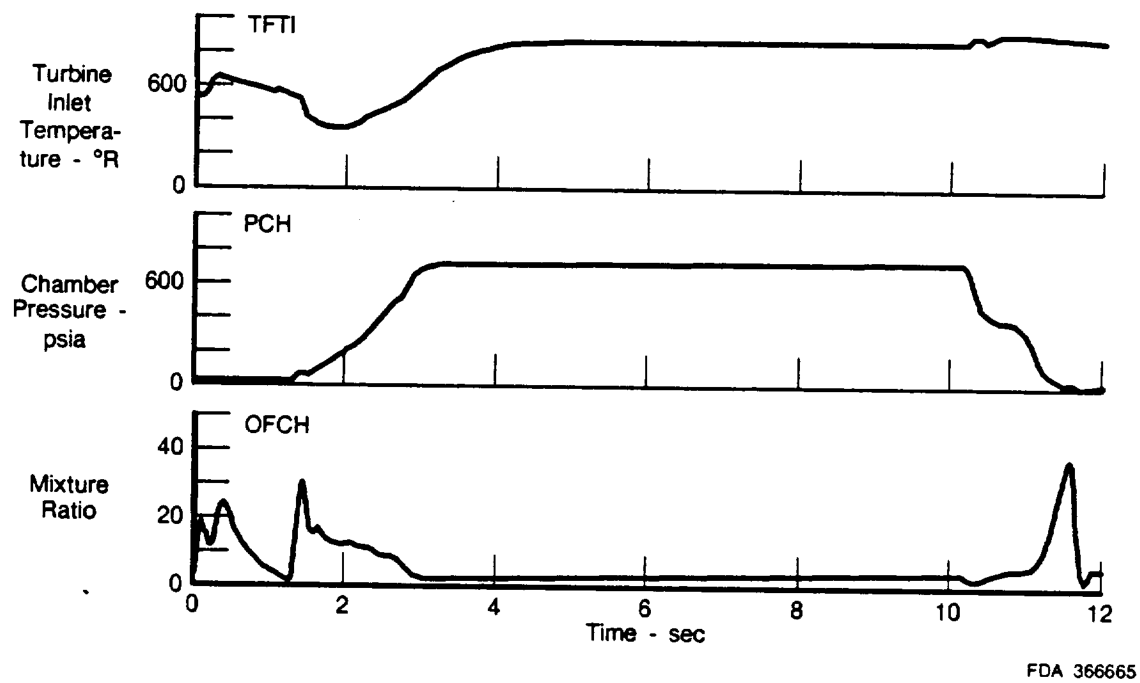


Figure 4.2.1.5-6. STBE Split Expander Operational Characteristics and Response During Start/Shutdown

4.2.1.5.5.4 Main Stage

The MOV and JBV are opened to maximum areas for main stage operation.

Thrust and main chamber mixture ratio are trimmed prior to flight during ground calibration testing by position trimming of the TBV and MOV control valves. Both the TBV and MOV trim stops are adjusted at the 706,500-pound vacuum thrust setting.

Analytical studies have shown that the ±3% thrust and mixture ratio requirement can be met with open-loop control.

4.2.1.5.5.5 Shutdown

The engine shutdown is initiated by placing the turbine bypass valve in its maximum area position when the shutdown signal is received from the vehicle (See Figures 4.2.1.5-4 through -6). This reduces turbopump turbine power and decelerates the turbopumps. Subsequently, the nozzle shutoff valve is closed, while the JBV and MOV closes due to a decrease in actuation ΔP's. The MOV must close last in the shutdown sequence to prevent unloading of the LO_2 pump.

4.2.1.6 Engine Configuration and Integration

4.2.1.6.1 Derivative STBE Split Expander Engine Assembly

The side and top views of the Derivative STBE Split Expander Engine assembly are shown in Figures 4.2.1.6-1 and -2, respectively.

4.2.1.6.2 GO_2 Heat Exchanger

The GO_2 heat exchanger provides gaseous oxygen to the oxygen tank for tank pressurization. The GO_2 heat exchanger uses oxidizer turbine exhaust duct flow as the heat source to vaporize the liquid oxygen as shown in Figure 4.2.1.2-1. The heat exchanger consists of five Haynes 214 stainless steel tubes wrapped in parallel around the exhaust duct. The exhaust duct wall is made of beryllium copper with trip-strip roughened walls to enhance the heat transfer. The tubes are packed in powdered copper to structurally isolate the tubes from the duct wall, while providing a good heat transfer medium. This design eliminates the possibility of accidental mixing of the oxygen and exhaust turbine flows, thereby eliminating a category 1 failure mode.

The GO_2 heat exchanger requires five 3/8-inch diameter tubes 41.0-feet long, wrapped around the 7-inch duct. The tubes have 0.015-inch thick walls and are separated from one another by 0.050-inch, requiring a total duct length of 3.70 feet. Figure 4.2.1.6-3 diagramatically presents the GO_2 heat exchanger geometry. The GO_2 heat exchanger has been thermally analyzed for the STBE 100 percent engine operating point with an oxygen flow rate of 8.4 lbm/sec. The heat exchanger is designed to supply 400 R oxygen to the tank. Figure 4.2.1.6-3 also summarizes the predicted heat exchanger performance.

4.2.1.6.3 Reliability, Maintainability, and Safety

4.2.1.6.3.1 Reliability

This section provides a complete preliminary Failure Modes and Effects Analysis (FMEA) for the Space Transportation Booster Engine (STBE) Split Expander Cycle. The section includes the definitions and details used to perform the analysis.

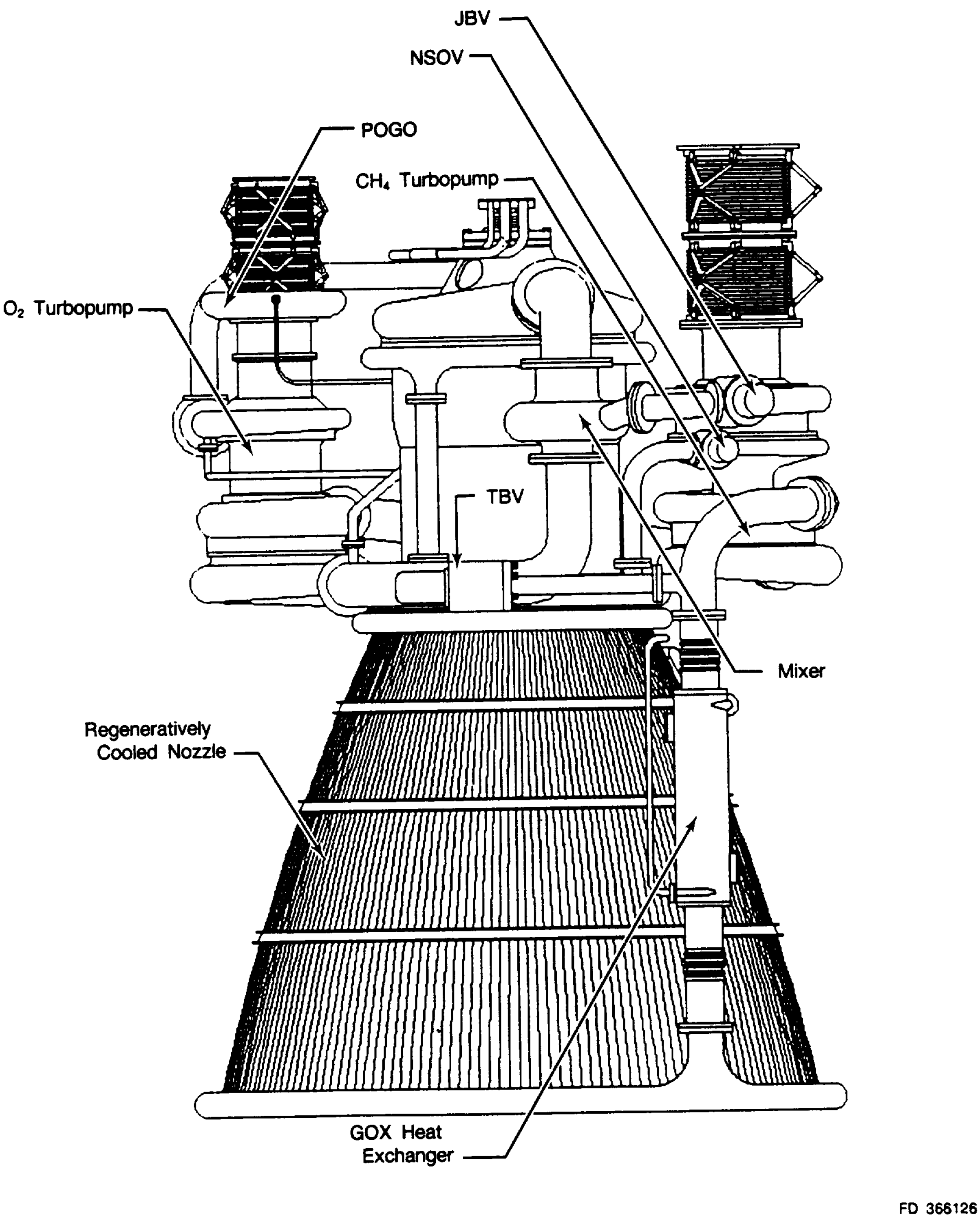


Figure 4.2.1.6-1. STBE Split Expander Engine Assembly — Side View

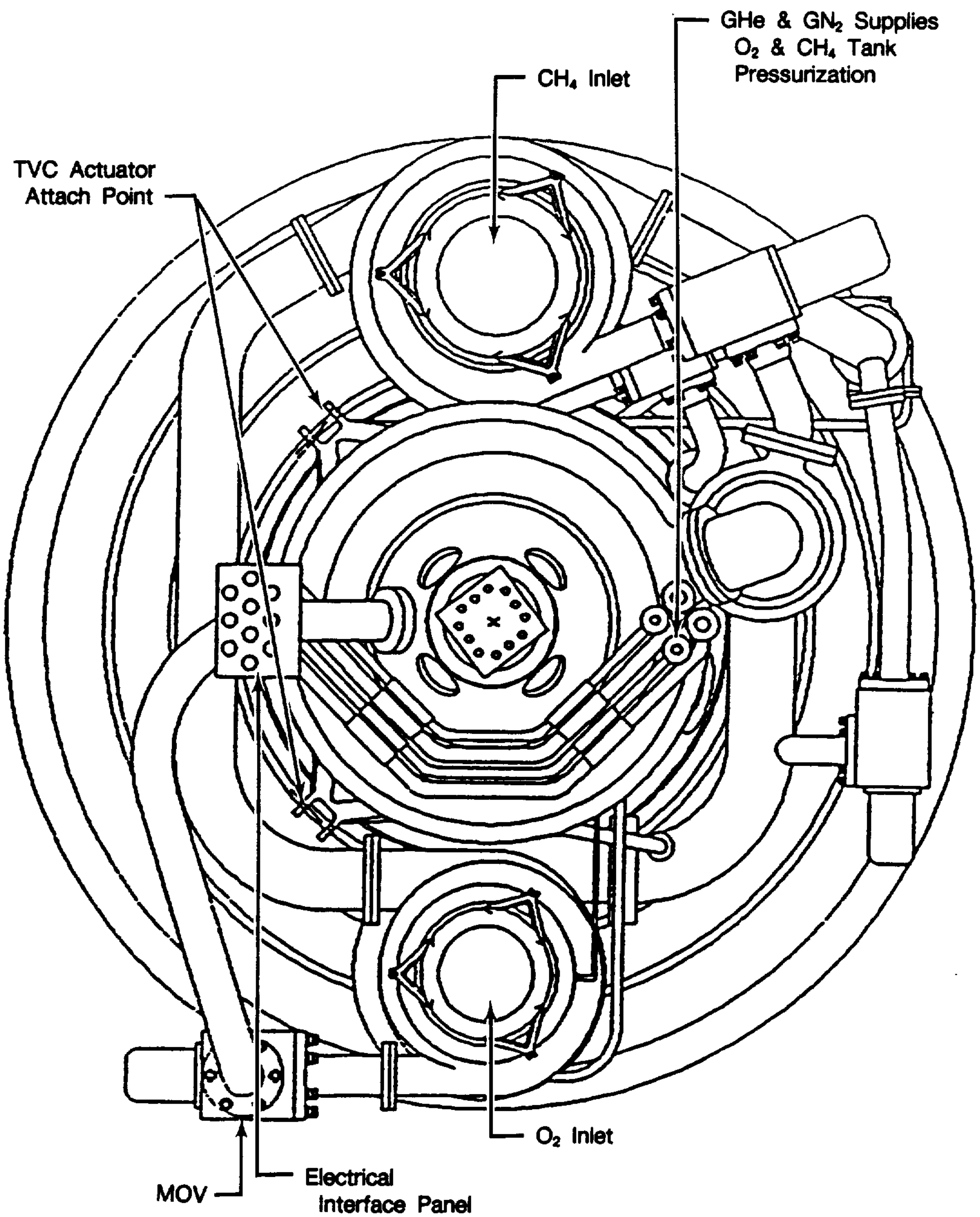


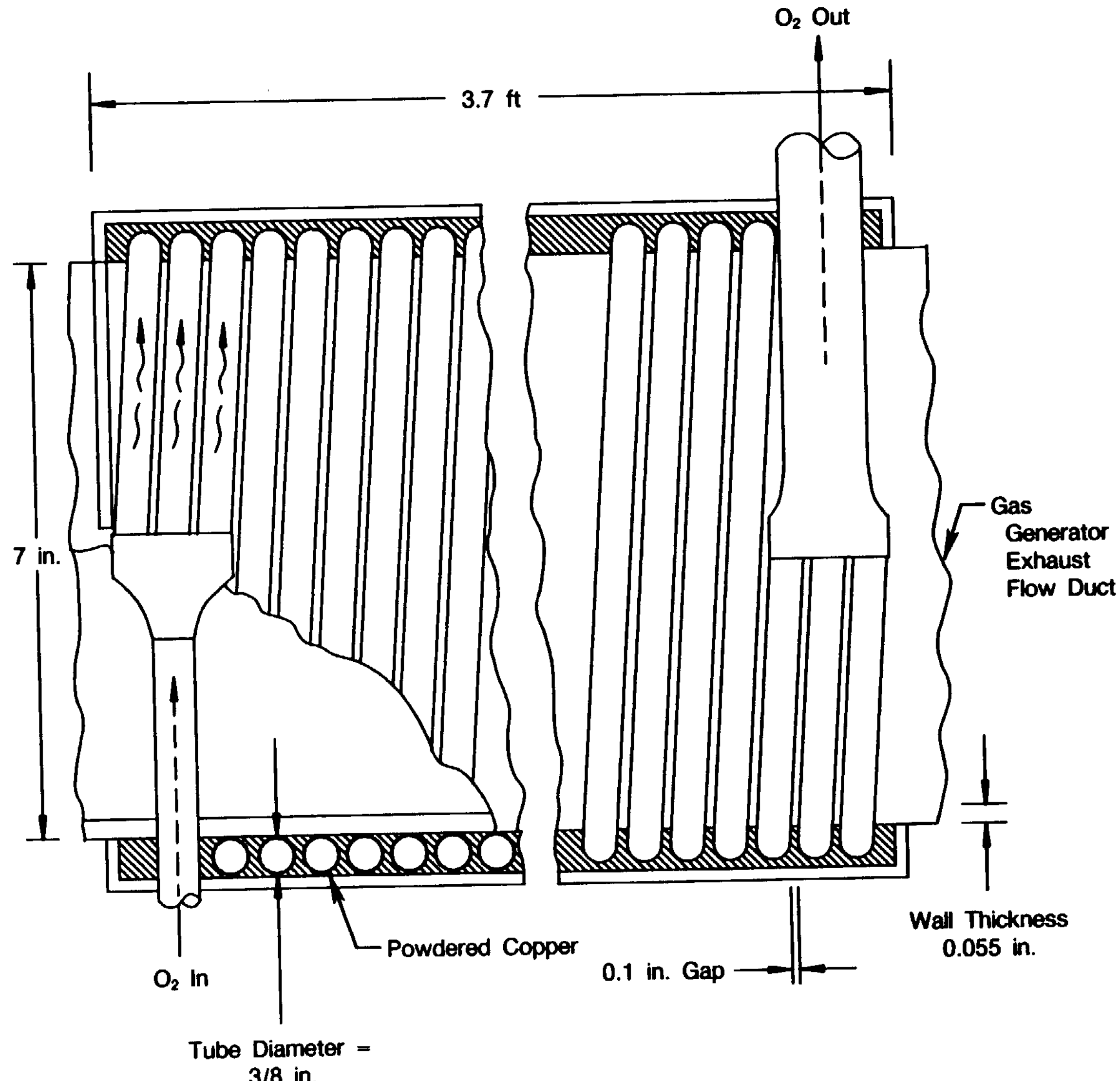
Figure 4.2.1.6-2. STBE Split Expander Engine Assembly — Top View

Introduction

Failure Modes and Effects Analysis (FMEA) for the STBE Split Expander Cycle Engine has been prepared to identify those items that are essential to engine operation. Engine components were analyzed to identify potential failure modes, determine their effect on engine operation, and rank the effects according to Condition Classification. The complete FMEA's are presented in Figure 4.2.1.6-4 and Table 4.2.1.6-1.

STBE GO_2 Heat Exchanger
Thermal and Flow Performance

Coolant Exit Duct	
Thrust - %	100
Inlet Temperature - R	772
Exit Temperature - R	766
Inlet Pressure - psia	960
Exit Pressure - psia	953
Flow Rate - lbm/sec	244
Oxygen	
Flow Rate - lbm/sec	8.7
Inlet Temperature - R	170
Exit Temperature - R	400
Inlet Pressure - psia	1224
Exit Pressure - psia	1067



GO_2 Heat Exchanger Length = 3.7 ft

Figure 4.2.1.6-3. STBE Derivative Split Expander GO_2 HEX Geometry and Performance Data

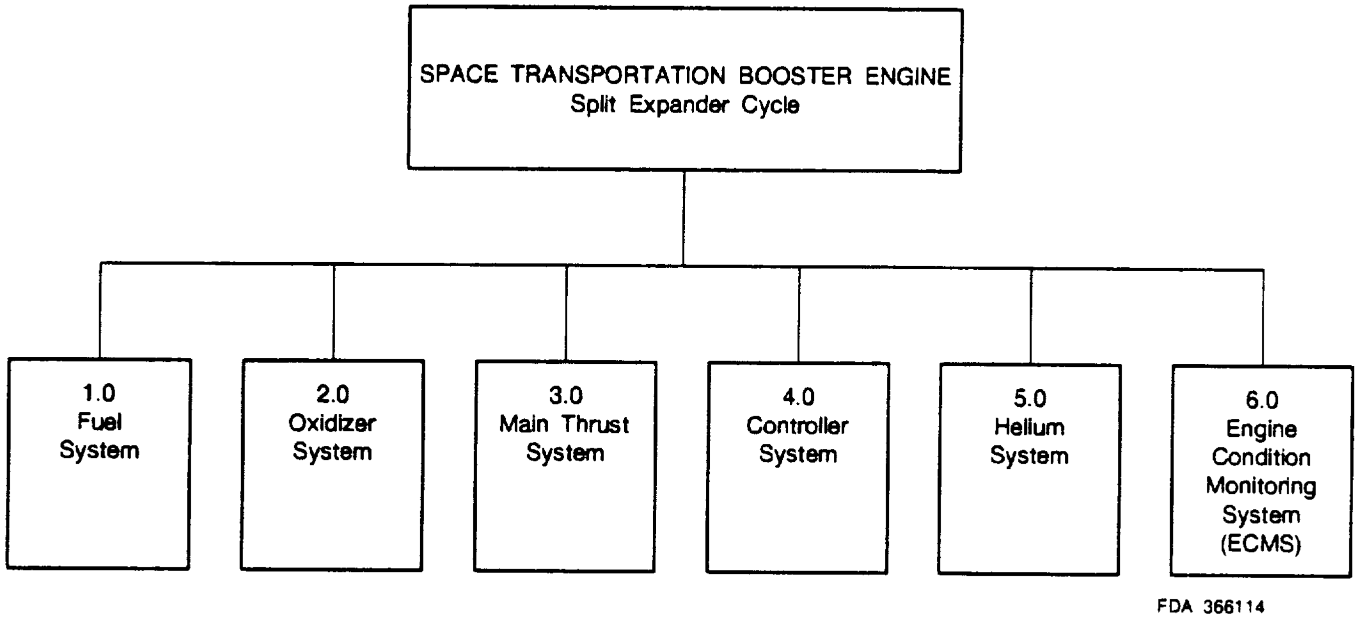


Figure 4.2.1.6-4. Preliminary FMEA — Major Engine Sections (Sheet 1 of 4)

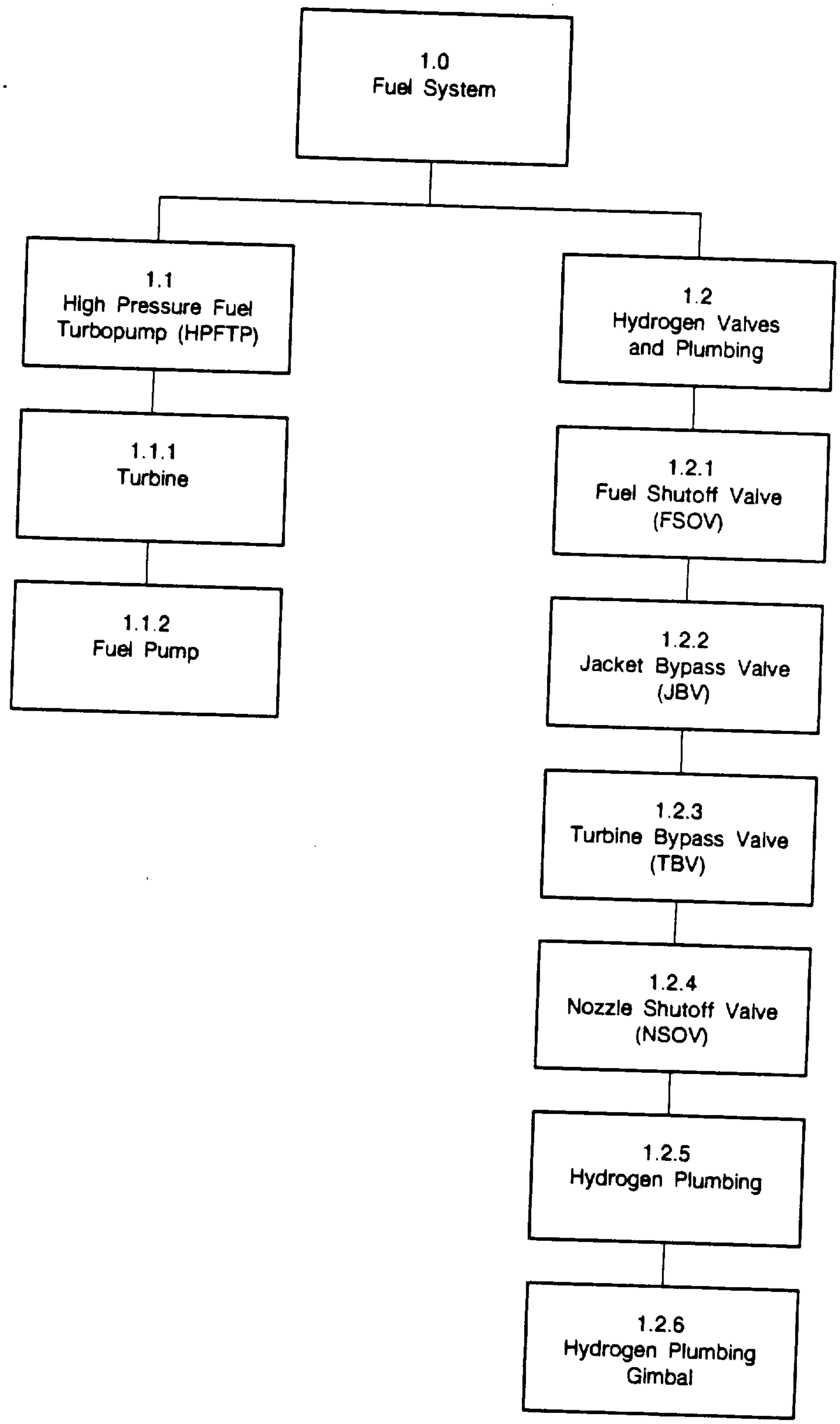


Figure 4.2.1.6-4. Preliminary FMEA — Fuel System (Sheet 2 of 4)

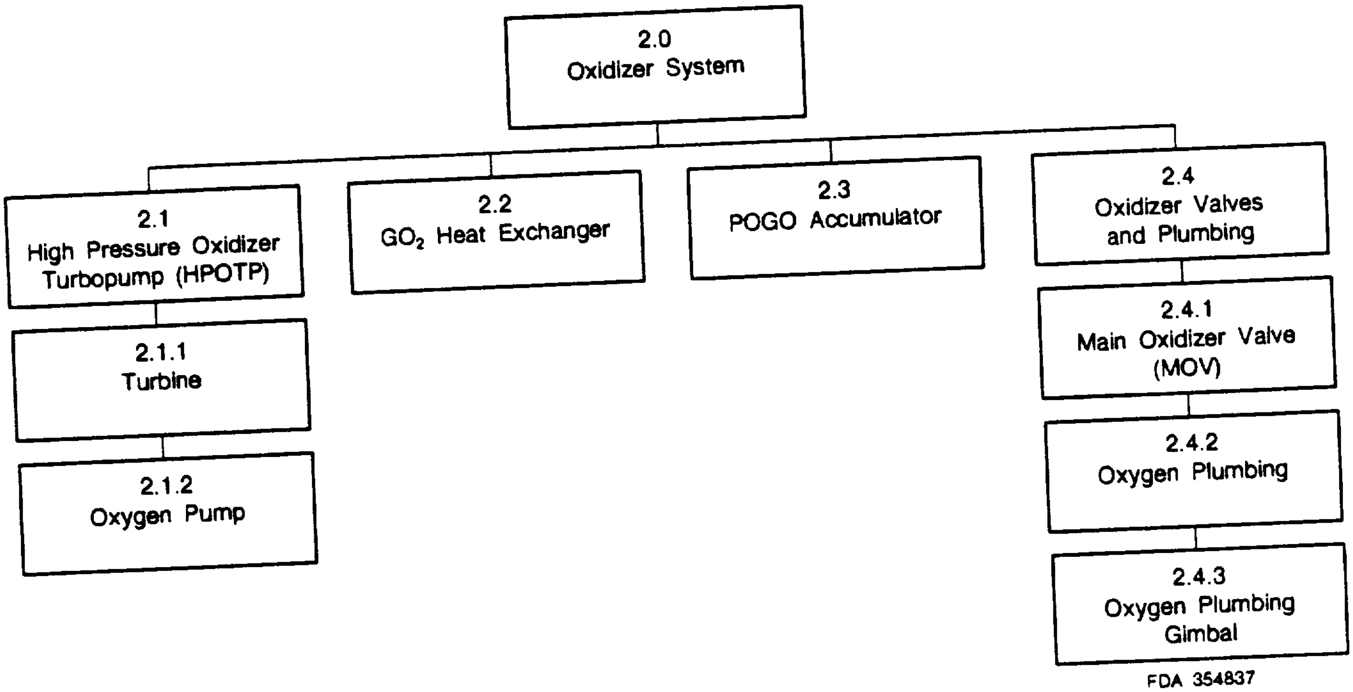


Figure 4.2.1.6-4. Preliminary FMEA — Oxidizer System (Sheet 3 of 4)

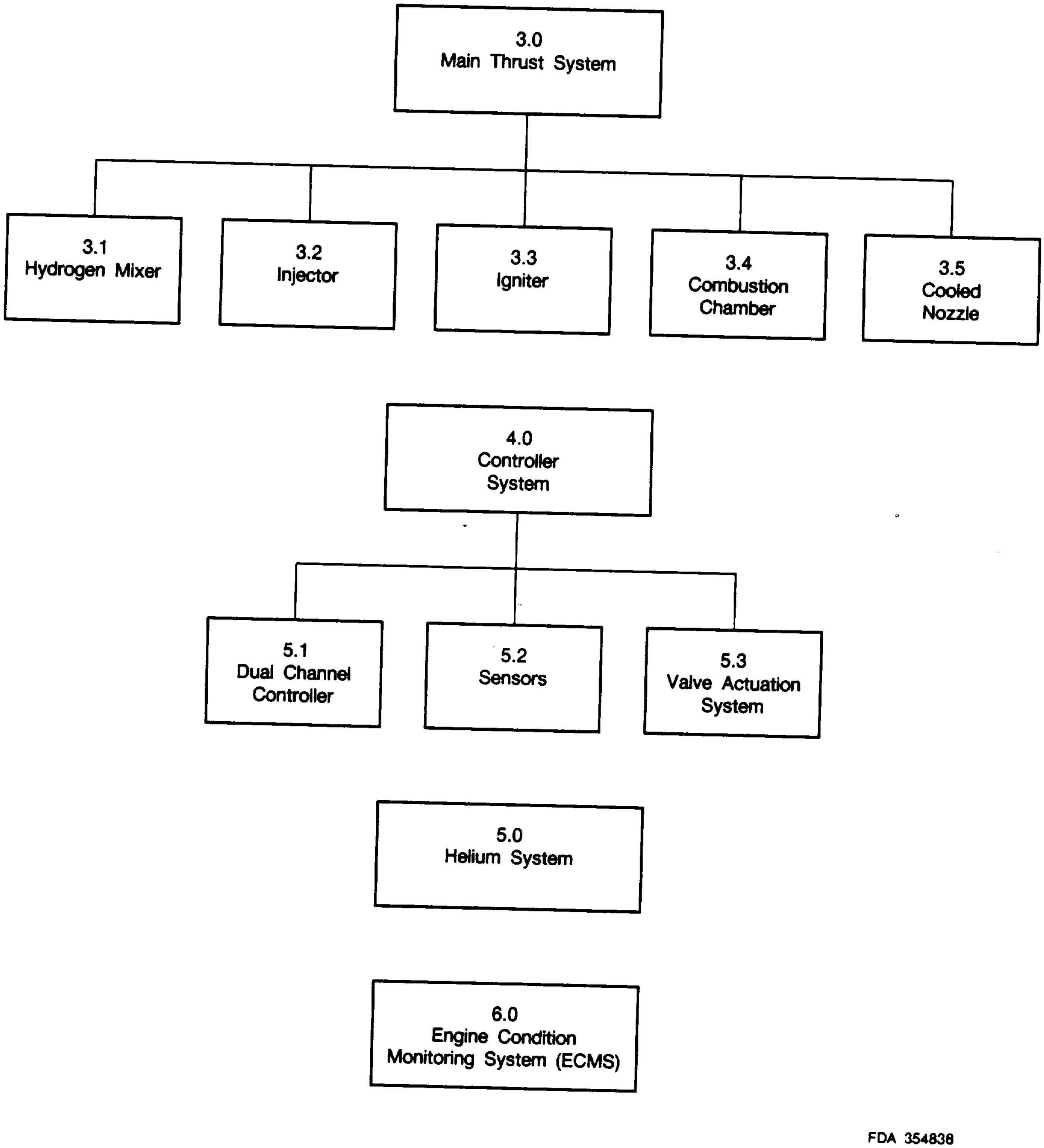


Figure 4.2.1.6-4. Preliminary FMEA — Main Thrust, Control, Helium, and Engine Condition Monitoring Systems (Sheet 4 of 4)

Table 4.2.1.6-1. *Failure Mode and Effects Analysis*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Fuel System Item 1.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 1 of 11
Issue Date: April 5, 1987
Rev. Date: Mar. 3, 1989

Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
1.0 FUEL SYSTEM Provides adequate pressure fuel for engine cooling requirements and main thrust chamber injection using expanding fuel from the nozzle jacket to power the turbine.				
1.1 HIGH PRESSURE FUEL TURBO-PUMP (HPFTP) Provides high pressurize liquid fuel to satisfy engine fuel and cooling requirements. Failure mode:				
1.1.1 TURBINE Uses expanding fuel from the nozzle jacket to create rotary motion to power the fuel pump. Failure mode: 1. Loss of turbine drive Cause: 1. Turbine blade or vane failure from fatigue.	Pump shutdown leading to an engine shutdown	II		Built in containment precludes impact to surrounding area. Use of advanced materials and cooling techniques provide enhanced turbine durability.
1.1.2 FUEL PUMP Pressurizes fuel for delivery to: The main thrust chamber and nozzle jacket. Failure mode: 1. Ruptured housing Cause: 1. Excessive load or material defect	Could lead to fire.	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
Failure mode: 2. Leakage through flange joint. Cause: 2. Seal, bolt, or flange failure such as fatigue.	Same as 1.	I		Same as 1.
Failure mode: 3. Bearing failure Cause: 3. Loss of cooling, excessive load, or defect.	Loss of bearing could lead to explosion.	I		Advanced bearing technology and controlled cooling coupled with inspection should preclude failures.

Table 4.2.1.6-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Fuel System Item 1.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
Page: 2 of 11
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
1.2 FUEL VALVES/PLUMBING Delivers and controls fuel flow from the tank to both the nozzle jacket and thrust chamber injectors.				
1.2.1 FUEL SHUT OFF VALVE (FSOV) Located downstream of fuel turbopump. Strictly an on/off functioning valve. Valve closed during engine pre-start cooldown, open during engine run, and closed at shutdown. Failure mode:				
Failure mode: 1. Fails open Cause: 1. Contamination, wear, loss of signal, or vibration.	1. Normal position at normal power level. No effect on engine operation.	III		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
Failure mode: 2. Fails closed Cause: 2. Same as above	2. Combustion chamber burn through leading to an engine shutdown.	I		Automatic shutdown would be controlled by redline limits
1.2.2 JACKET BYPASS VALVE (JBV) Bypasses fuel flow from the cooling jacket directly to the combustion chamber.				
Failure mode: 1. Fails open Cause: 1. Contamination, wear, loss of signal, or vibration.	1. Reduced cooling flow to nozzle jacket. Possible nozzle burn through and loss of engine performance.	III		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
Failure mode: 2. Fails closed Cause: 2. Same as above	2. Eliminates fuel flow directly to the main combustion chamber. Possible loss of engine performance.	II		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
1.2.3 TURBINE BYPASS VALVE (TBV) Bypasses hot fuel from the turbines to control pump turbine speed.				
Failure mode: 1. Fails open Cause: 1. Contamination, wear, loss of signal, or vibration.	1. Reduced turbine speed resulting in engine performance loss.	II		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.

Table 4.2.1.6-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Fuel System Item 1.0
Prepared by: R.L. Pugh
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
Failure mode: 2. Fails full closed. Cause: 2. Same as above	2. Unable to control turbine speed. Possible shutdown.	II		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
1.2.4 NOZZLE SHUT OFF VALVE (NSOV) Controls fuel flow to nozzle cooling jacket.				
Failure mode: 1. Fails open Cause: 1. Contamination, wear, loss of signal, or vibration.	1. No effect on normal engine operation at NPL.	III		Material selection will reduce wear. Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
Failure mode: 2. Fails closed Cause: 2. Same as above	2. No cooling flow to nozzle or hot fuel flow to power pump turbines resulting in an engine shutdown.	II		Automatic shutdown would be controlled by redline limits.
1.2.5 FUEL SYSTEM PLUMBING Plumbing transports the fuel to and from the valves to the designated areas.				
Failure mode: Fatigue Cause: Vibration, thermal growth, or material defect.	Possible fuel leak and fire.	I		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems. Proper bracketry provided to dampen line vibration.
1.2.6 FUEL PLUMBING GIMBAL Provides mechanical interface between the HPFTP inlet with the vehicle main fuel supply. Permits engine vectoring without distortion of the fuel plumbing or disrupting fuel flow.				
Failure mode: Fatigue Cause: Vibration, thermal growth, or material defect.	Possible fuel leak and fire.	I		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems.

Table 4.2.1.6-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBI Split Expander Cycle
Functional Assy: Oxidizer System Item 2.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
2.0 OXIDIZER SYSTEM				
Provides oxygen at an adequate pressure to enter the main oxygen pump.				
2.1 HIGH PRESSURE OXIDIZER TURBOPUMP (HPOTP)				
Provides high pressure oxygen to be injected and combusted in the main thrust chamber.				
2.1.1 TURBINE				
Uses expanding fuel from the nozzle jacket to create rotary motion to power the oxygen pump.				
Failure mode: 1. Loss of turbine drive	Pump shutdown leading to an engine shutdown	II		Built in containment precludes impact to surrounding area. Use of advanced materials and cooling techniques provide enhanced turbine durability.
Cause: 1. Turbine blade or vane failure from fatigue.				
2.1.2 OXYGEN PUMP				
Pressurizes liquid oxygen for delivery to the GOX heat exchanger (to pressurize O2 tank and POGO accumulator) and the main thrust chamber.				
Failure mode: 1. Ruptured housing	1. Oxygen leak could lead to explosion	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
Cause: 1. Excessive load or material defect				
Failure mode: 2. Leakage through flange joint.	2. Same as 1.	I		Same as 1.
Cause: 2. Seal, bolt, or flange failure such as fatigue.				
Failure mode: 3. Bearing failure	3. Loss of bearing could cause an explosion.	I		Advanced bearing technology and controlled cooling coupled with inspection should preclude failures.
Cause: 3. Loss of cooling, excessive load, or defect.				

Table 4.2.1.6-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Oxidizer System Item 2.0
Prepared by: R.L. Pugh
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
2.2 GOX HEAT EXCHANGER The GOX heat exchanger uses the O2 pump turbine exhaust to vaporize a portion of the O2 pump discharge flow to pressurize the oxygen tank.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or defect	1. Possible leakage of gaseous oxygen which may cause an explosion.	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
Failure mode: 2. Plug Cause: 2. Contamination	2. Loss of tank pressure supply. Reduced LOX inlet pressure may cause loss of engine performance.	II		Design of HEX precludes any mixing of fuel and GOX due to structural failure. Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature.
2.3 POGO ACCUMULATOR The POGO Accumulator eliminates transmission of low frequency flow oscillations in to the high pressure oxidizer turbopump which in turn eliminates main combustion chamber pressure oscillations.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or defect	Possible leakage of gaseous oxygen which may cause an explosion.	I		Design philosophy to provide margins of 1.5 or better in materials and tight inspection criteria should preclude failures of this nature. DVS testing to assure proper design margins.
2.4 OXIDIZER VALVES/PLUMBING Delivers and controls the oxygen flow from the tank to the main thrust chamber injectors.				

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Table 4.2.1.6-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Oxidizer System Item 2.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criti-cality	Detection Method	Considerations
2.4.1 MAIN OXIDIZER VALVE (MOV) The MOV is a dual purpose butterfly valve located between the oxygen turbopump and the main chamber injector. The MOV serves as an on/off valve during start and shutdown and as a scheduled control valve during engine operation.				
Failure mode: 1. Fails open Cause: 1. Contamination, wear,loss of signal, or vibration.	1. Normal position at normal power level. No effect on engine operation.	III		Material selection will reduce wear *Selected valve design will lessen the effect of contaminants. DVS and bench qualification testing to assure proper design margins.
Failure mode: 2. Fails closed Cause: 2. Same as above.	2. Abnormal engine shutdown	II		Same as above
2.4.2 OXYGEN PLUMBING Plumbing transports the various propellants to and from the valves to the designated areas.				
Failure mode: Fatigue Cause: Vibration, thermal growth, or material defect.	Possible fire.	I		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems.
2.4.3 OXYGEN PLUMBING GIMBAL Provides mechanical interface between the HPOTP inlet with the vehicle oxygen supply. Permits engine vectoring without distortion of the oxygen plumbing or disrupting fuel flow.				
Failure mode: Fatigue Cause: Vibration, thermal growth, or material defect.	Possible LOX leak and explosion.	I		The design technology of plumbing and seals coupled with material selection and Quality system all go to preclude problems. DVS testing to assure proper design margins.

Table 4.2.1.6-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Main Thrust System Item 3.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
3.0 MAIN THRUST SYSTEM Injects, atomizes, mixes, and burns fuel and oxygen to form hot gaseous reaction products, which are accelerated through the chamber throat and expanded through the chamber nozzle to be ejected at a high velocity to create thrust. The chamber also provides a heat exchanger for the fuel turbopump operation.				
3.1 FUEL MIXER Mixes hot fuel gas from the drive turbines with high pressure fuel from the pump. Forms a mixture suitable for injection into the main chamber. Failure mode: Cause:				
3.2 INJECTOR Meters and injects the chamber propellants into the combustion chamber, causing atomization and mixing for efficient combustion.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or material defect.	Premature mixing of fuel and LOX could lead to an explosion.	1		Structural margins of 1.5 inherent in the design should preclude this type of failure. Elimination of braze and weld will preclude premature mixing failures.
Failure mode: 2. Loss of liner integrity Cause: 2. Loss of cooling to liner	Burn through could lead to engine shutdown	1		Design of Rigimesh cooling and structural margins should preclude problems.
3.3 IGNITER A series of sparks across the igniter plug gap and the addition of the proper igniter propellant flow creates a continuous torch which lights and keeps lit the chamber propellants.				
Failure mode: 1. Shorting Cause: 1. Cracked pressure insulation	1. No effect on normal engine operation. Engine unable to start or restart.	IIIR		Redundant spark igniters are provided

Table 4.2.1.6-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Main Thrust System Item 3.0
Prepared by: R.L. Pugh
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
Failure mode: 2. Loss of electrical Cause: 2. Circuit malfunction	2. Same	II		Redundant electrical leads are provided.
Failure mode: 3. Loss of propellant flow Cause: 3. Blockage of passage.	3. Same	II		Filters provided in lines. preclude loosing enough to create a problem.
3.4 COMBUSTION CHAMBER[4] Provides a chamber for the mixing, atomization, burning and partial expansion of the chamber propellants, accelerating the hot gases to an area ratio of 2.0. Provides a heat exchanger for delivery of hot fuel to the turbomachinery.				
Failure mode: 1. Non-uniform temperature profile. Cause. 1. Malfunction of the injector.	1. Burn through could lead to engine shutdown	II		Automatic shutdown would be controlled by redline limits.
Failure mode: 2. Loss of propellant flow Cause: 2. Plugged propellant passages.	2. Fuel line blockage leads to LOX rich and overtemp with possible explosion. LOX plug would lead to partial loss of performance, and excessive plugging would lead to engine shutdown.	I, II		Design features adequate filters, and the numbers of injectors should
3.5 COOLED NOZZLE Primary purpose is to increase engine performance by expanding the combustion chamber gases from an area ratio of 2.0 to an area ratio of 28.0. Also acts as a heat exchanger to expand the fuel for the turbine drives.				
Failure mode: 1. Structural failure Cause: 1. Vibration, thermal growth, or material defect.	1. Leakage of fuel into nozzle or external to nozzle could lead to fire and LOX rich combustion burn through in chamber	I.II		Structural margins of 1.5 inherent in the design should preclude this type of failure.
Failure mode: 2. Loss of propellant due to tube failure Cause: 2. Vibration, thermal growth, or material defect.	2. LOX rich combustion leading to chamber burn through resulting in engine shutdown.	II		Automatic shutdown controlled by redline limits.

Table 4.2.1.6-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Control System Item 4.0
Prepared by: R.L. Pugh
Approved by: W.E. Annas
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
4.0 CONTROLLER SYSTEM Provides a self-contained system for engine control. System includes and electronic controller, engine sensors, actuation devices, valves, and spark igniters. Failure mode: Cause:				
4.1 DUAL CHANNEL CONTROLLER Provides closed loop engine control and monitoring using input from sensing devices and output to actuation devices. Failure mode: Loss of multiplexing capability converting sensor analog or frequency signals to digital signals. Cause: Internal failure of converter circuit.	No effect on normal engine operation.	IIIR		Dual redundant controller allows normal operation after the first failure.
4.2 SENSORS TBD Failure mode: Cause:				
4.3 VALVE ACTUATION SYSTEM TBD Failure mode: Cause:				

Table 4.2.1.6-1. *Failure Mode and Effects Analysis (Continued)*

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Helium System Item 5.0
Prepared by: R.L. Pugh
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criticality	Detection Method	Considerations
5.0 HELIUM SYSTEM Delivers and controls helium flow from the vehicle supply to be used for valve operation, emergency shutdown operation, and turbopump spin assist. TBD				

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Table 4.2.1.6-1. Failure Mode and Effects Analysis (Continued)

FAILURE MODE AND EFFECTS ANALYSIS

End Item: STBE Split Expander Cycle
Functional Assy: Eng. Condition Monitor Sys. Item 6
Prepared by: R.L. Pugh
Approved by: W.E. Annas
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Item No., Function, Failure Mode and Cause	Failure Effect on System	Criti-cality	Detection Method	Considerations
6.0 ENGINE CONDITION MONITORING SYSTEM (ECMS) TBD				

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Results

This section is intentionally left blank at this time. It will be developed as the analysis proceeds.

Conclusions

This section is intentionally left blank at this time. It will be developed as the analysis proceeds.

Applicable Documents

NHB 5300.4(1D.2) Safety, Reliability and Quality Provisions for SSME Programs

Procedure

This report was prepared in accordance with STBE Reliability requirements.

Ground Rules and Assumptions

The following ground rules and assumptions were used in the preparation of the FMEA.

Level of Analysis

a. The analysis is conducted at the component and major subassembly level. In subsequent updates, the FMEA will contain both a hardware and a functional analysis. To show the distinction, the index numbers have been modified to differentiate between functional and hardware type of analysis.

b. Condition category I and II items will be analyzed to the level necessary to verify adequate controls are in place.

c. External fire, explosion, or case penetration that could endanger the remaining engines are classified as Condition Classification I.

d. The worst case effect of leakage is fire/explosion. In this analysis, leakage will be classified as Condition I.

e. The analysis was conducted considering the engine operation at normal power. Subsequent updates will consider the mission phases in the following paragraph.

f. The helium solenoids, valve actuators, sensors, and monitoring devices have not been analyzed. Analysis of these items will be provided in subsequent updates.

Mission Phases

The engine will be analyzed for potential failure modes of a single flight in each of the following mission phases:

Event	*Phase*	*Abbreviation*
Pre-start	Pre-ignition	P
Start Command	Engine Start	S
Normal Power	Main Stage Operation	N
Max Thrust	Main Stage (Lift Off)	M
Cutoff Command	Shutdown	C
Dump	After Shutdown	D

Failure Modes

a. Failure modes will be identified for each level or output applicable to the operational phase being considered.

b. The analysis will consider only one failure mode to have occurred at any given time and will be the basis for establishing Condition Classification.

c. Leakage at all mechanical joints shall be analyzed.

d. Welded or brazed joints shall be analyzed for structural failure.

e. Failure mode causes shall be identified for all Condition Classifications I, II, and III.

Reaction Time

The analysis will determine the time for the failure effect to occur, and it is specified in units of time as indicated below:

	Definition	*Abbreviation*
Immediate	— Less than a second	IMM
Seconds	— 1 to 60 seconds	SEC
Minutes	— 60 seconds to 60 minutes	MIN
Hours	— 60 minutes to 24 hours	HRS
Days	— 24 hours to mission completion	DYS

If a failure detection method is available, it is specified with time to safely correct the problem. If a detection system is available, but would not safely correct the problem, this is also noted.

Failure Effects

a. Failure effects will be analyzed for each identified failure mode. Where a piece part failure can cause a failure of another part, the Condition Classification will be based on the likely effect of the resultant or combined failures.

b. Condition designation should reflect "the most likely" potential effect of the failure mode in either countdown or flight. This includes possible catastrophic effects, such as fire/explosion, as well as effects of loss of hardware functions. Single failures, such as leakage of LO_2, in the presence of a possible ignition source, will be listed as potential fire/single failure point. Leakage of hot gas is classified Condition I.

Structural Failure Modes

Structures are excluded from the FMEA, with the exemptions listed below.

a. Pressure vessels, components housings, ducts, fluid lines, sliding joints, expansion joints, bolts, attach fittings, or load carrying members such as rods will be analyzed for structural failures.

b. Structural failures of piece parts shall be considered valid failure causes for component failure mode analysis.

c. Items which have a single mechanical barrier between oxidizer and fuel/combustible gas.

d. Items that are known to develop "acceptable defects" within their allowed time for usage, shall be analyzed for worst case of defect propagation.

e. Aerodynamically sensitive items.

f. Items having internal cavities which can induce an internal overpressure from migrating fluid because of leak from inside or outside.

g. Leakage at all joints that are formed by weld or braze shall be analyzed to assess the effect of a leak impinging on other components or flammable surfaces.

h. Welds or braze joints that cannot be inspected will be analyzed for leakage and for structural failure effects.

Criticality Category

The criticality category for each failure mode will be assessed for its effect on missions as follows:

Condition	*Mission Effect*
I	A potential failure mode resulting in fire/explosion or other hazardous condition that could impact the surrounding area.
II	A potential failure mode that could result in an unscheduled safe engine shutdown.
III	A potential failure mode that could result in the engine safely operating outside of required parameters.

4.2.1.6.3.2 Maintainability

Preliminary maintainability design criteria for the STBE has been defined and provided to design engineers in a memorandum. The design criteria was derived from the statement of work (SOW), preliminary guidance from ALS airframers, and experience gained from other liquid rocket programs. Experience gained includes the Pratt & Whitney RL10 and Alternate Turbopump Development (ATD) programs and information from various NASA reports relative

to the SSME, F1, J2, H1, RS-27, Thor, and Atlas programs. Updates will be made to the maintainability design criteria as additional requirements are identified.

Pratt & Whitney maintainability engineering has been working in conjunction with the ALS airframers to define an overall maintenance concept for the STBE. Definition of the maintenance concept will provide necessary guidance in identifying those propulsion system components that are either line replaceable units (LRU's) or modules. The definition and lists of LRU's and modules and preliminary maintenance concepts will be provided in subsequent reports.

4.2.1.6.3.3 System Safety

To support the development of design requirements, System Safety developed Fault Tree Analyses of the split expander engine systems and their major components. These Fault Trees are high-level models to study the overall systemic effects of "generic" events such as "turbopump mechanical malfunction". Detailed fault trees investigating events such as "bearing rate fracture" within a turbopump will be developed during Phase B studies.

The Fault Trees were analyzed to identify those events with possible catastrophic results. The identified events and their effects on the system were then analyzed to determine safety requirements which would eliminate or reduce the probability and/or severity of the undesired effect. These requirements have been summarized and provided to Project Engineering for inclusion as engine system design requirements during Phase B design activities.

The objective of this effort is to reduce the probability of a catastrophic engine event (one that results in the loss of the payload or the vehicle, or the death or serious injury of a person) to the lowest possible level. This will be accomplished by using the fault trees to identify those diagnostic elements which detect potentially hazardous conditions in time to effect an engine shutdown before the event becomes catastrophic. The overall goal is to contain the damage within the malfunctioning engine thus avoiding potential damage to an adjacent engine.

4.2.1.6.4 Engine Performance

The modified STME nozzle performance was determined during the preliminary design using the accepted JANNAF methodolgy. Table 4.2.1.6-2 lists the detailed analysis for the design power level (DPL) of 600,000 lbf sea level thrust.

Table 4.2.1.6-2. *Modified STME Split Expander Nozzle Performance*

Design Power Level	
Pressure — psia	733.7
Mixture Ratio	3.5
Area Ratio	13.5
Ideal Isp — sec	340.6
Delta Isp ERE — sec	−6.86
Delta Isp KIN — sec	−0.50
Delta Isp TDK — sec	−4.22
Delta Isp BLM — sec	−1.29
Delta Isp Vac — sec	327.7
Flowrate — lbm/s	2,155.8
Vac. Thrust	706,500.0

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During this study program, detailed aerothermal analyses were made to predict component performance levels and these were incorporated into a steady-state model of the complete engine.

Simplified flow schematics are presented in Figure 4.2.1.6-5 with key operating parameters noted for the design thrust level. Table 4.2.1.6-3 defines performance of the individual components and their operating environments for the modified STME at the design power level.

The modified STME uses an external GO_2 heat exchanger to pressurize the LO_2 tank. This eliminates a category 1 failure mode. The heat exchanger uses the hot methane jacket exit flow to vaporize a small amount of LO_2 which is returned to the tank. The methane tank is pressurized with gaseous methane tapped off at the exit of the second turbine that drives the oxidizer main pump.

The modified STME was analyzed for a single operating thrust of 600,000 lbf at sea level.

4.2.1.6.5 Engine Costs

This section summarizes cost estimates for the 600K SL thrust, 734 psia chamber pressure, Derivative STBE Split Expander cycle. Table 4.2.1.6-4 summarizes significant costs for the engine.

The DDT&E Cost includes all of the functions required to design, develop, test and evaluate the engine system. All of the DDT&E functions shown in the ALS engine WBS (see Volume III) have been included. Development Cost is based on a 90-month phase C/D program with 960 engine firings for the STME Split Expander and 488 for the Derivative STBE Split Expander. Sufficient accountable firings have been included in the program to demonstrate 0.99 engine reliability with one failure.

The engine Theoretical First Unit (TFU) production cost includes all the recurring operational production cost elements specified in the ALS engine WBS. It includes manufacturing and acceptance of the Integrated Engine System, System Engineering and Integration, Program Management, Facilities Maintenance and Tooling Maintenance. The TFU estimate is based on a lot size of 100 and a 90-percent learning curve.

The Operations Cost per launch per engine includes all costs associated with the operational flight program as described in the ALS engine WBS. It includes Program Management, System Engineering and Integration, Facilities Maintenance, Operation and Support, and Training. The Operations Cost is based on a flight rate of 10 missions per year and it is the estimated cost that will be achieved after 100 total missions have been flown.

4.2.2 Unique Split Expander Cycle Engine

4.2.2.1 Engine Design Evolution

The STBE LO_2/CH_4 Split Expander Engine Study was initiated during the second quarter of 1988 as a Normal Power Level (NPL) design at 625K lbf sea level thrust. This engine was discussed in FR-19691-3 including flow schematic and cycle description, and is shown in Figure 4.2.2.1-1.

Further engine study refined the design through the last quarter of 1988. The significant changes from the initial design included the elimination of low-pressure boost pumps and the increased thrust to 750K lbf sea level as the Design Power Level. The engine assembly and major characteristics are shown in Figure 4.2.2.1-2.

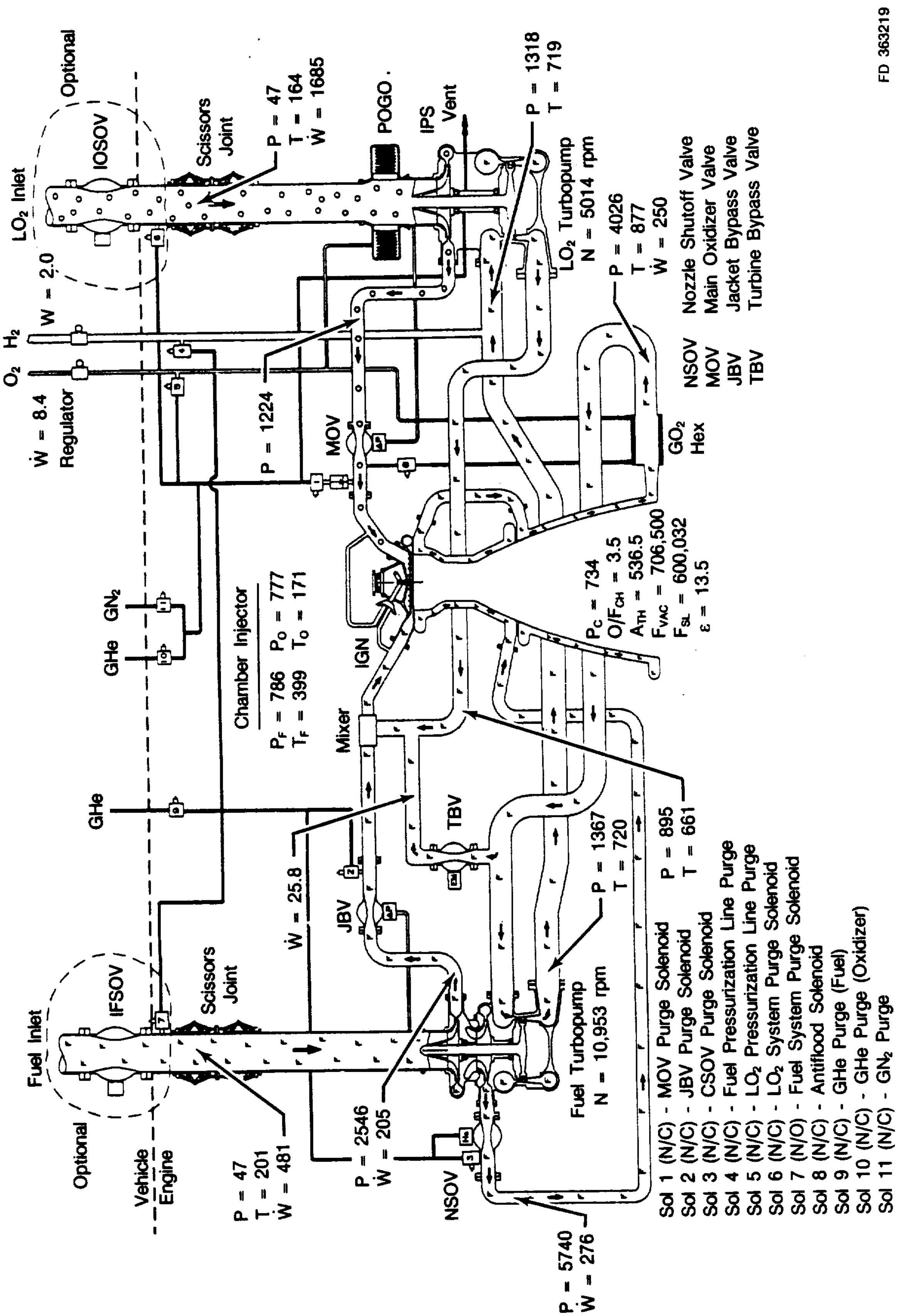

Figure 4.2.1.6-5. STBE Derivative Split Expander Cycle Engine Operating Characteristics at Normal Power Level

Table 4.2.1.6-3. *Modified STME Derivative Split Expander Engine Performance — Design Power Level*

PRATT & WHITNEY
STME LOX/CH4 MODIFIED ENGINE

ENGINE PERFORMANCE PARAMETERS
****

CHAMBER PRESSURE	PSIA	733.7
VAC ENGINE THRUST	LBF	706500.
S.L. ENGINE THRUST	LBF	600032.
ENGINE FLOW RATE	LBM/S	2155.8
DEL. VACUUM ISP	SEC	327.73
THROAT AREA	IN2	536.50
NOZZLE EXIT DIA.	IN	96.03
NOZZLE LENGTH	IN	93.39
NOZZLE AREA RATIO		13.5
ENGINE MIXTURE RATIO		3.50

HEAT TRANSFER/COMB PERFORMANCE
****

CHAMBER TEMPERATURE DEG R	6343.
CHAMBER GAS CONSTANT	71.
CHAMBER GAMMA	1.20
COMBUSTION C* EFFICIENCY	0.980
CHAMBER/NOZZLE COOLANT DP	1480.
CHAMBER/NOZZLE COOLANT DT	625.
CHAMBER/NOZZLE COOLANT Q	138510.
CHAMBER MATERIAL	HAYNES-230
NOZZLE MATERIAL	HAYNES-230
COOL CONFIG.	COUNTER & PARALLEL

ENGINE STATION CONDITIONS
****

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	201.0	481.1	123.1	26.40
1ST STAGE EXIT	2545.9	218.3	481.1	148.2	26.53
JBV INLET	2533.6	218.3	204.9	148.2	26.52
JBV EXIT	895.3	226.8	204.9	148.2	25.45
PUMP EXIT	5740.3	249.9	276.2	187.8	26.36
NSOV INLET	5740.3	249.9	276.2	187.8	26.36
NSOV EXIT	5581.7	250.9	276.2	187.8	26.27
COOLANT INLET	5506.4	251.3	276.2	187.8	26.23
COOLANT EXIT	4026.2	876.7	276.2	689.3	6.51
TBV INLET	3987.8	876.5	25.8	689.3	6.45
TBV EXIT	895.3	846.8	25.8	689.3	1.59
CH4 TRB INLET	3987.8	876.5	250.3	689.3	6.45
CH4 TRB EXIT	1381.9	720.6	250.3	597.3	2.95
CH4 TRB DIFFUSER	1367.1	720.3	250.3	597.3	2.92
LOX TRB INLET	1318.9	719.3	250.3	597.3	2.82
LOX TRB EXIT	939.7	670.6	250.3	571.5	2.18
LOX TRB DIFFUSER	918.8	670.0	250.3	571.5	2.13
CH4 TANK PRESS	47.0	641.7	2.0	571.5	0.11
GOX HEAT EXCH IN	918.8	670.0	248.3	571.5	2.13
GOX HEAT EXCH OUT	895.3	661.9	248.3	566.8	2.10
MIXER	895.3	408.4	479.1	394.4	4.74
FSOV INLET	832.9	403.1	479.1	394.4	4.43
FSOV EXIT	796.0	399.8	479.1	394.4	4.24
CHAMBER INJ	786.2	398.9	479.1	394.4	4.19
CHAMBER	733.7				

* OXYGEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	164.0	1685.1	61.6	70.98
PUMP EXIT	1223.8	168.9	1685.1	65.5	71.32
O2 TANK PRESS	47.0	400.0	8.4	204.4	0.36
OCV INLET	1198.0	169.0	1676.7	65.5	71.28
OCV EXIT	796.6	170.4	1676.7	65.5	70.63
CHAMBER INJ	777.3	170.5	1676.7	65.5	70.60
CHAMBER	733.7				

* VALVE DATA *

VALVE	DELTA P	AREA	FLOW	% BYPASS
JBV	1638.	1.47	204.90	42.59
NSOV	159.	6.39	276.16	
TBV	3093.	0.49	25.85	9.36
FSOV	37.		479.06	
OCV	401.	14.83	1676.70	

* INJECTOR DATA *

INJECTOR	DELTA P	AREA	FLOW	VELOCITY
FUEL	52.	53.45	479.06	281.61
LOX	44.	45.23	1676.70	75.61

Table 4.2.1.6-3. Modified STME Derivative Split Expander Engine Performance — Design Power Level (Continued)

PRATT & WHITNEY
STME LOX/CH4 MODIFIED ENGINE

****
* TURBOMACHINERY PERFORMANCE DATA *
****

TURBINE PERFORMANCE CHARACTERISTICS

		MAIN FUEL TURB. STAGE 1	MAIN FUEL TURB. STAGE 2	MAIN OXID. TURB. STAGE 1	MAIN OXID. TURB. STAGE 2
EFFICIENCY	(T/T)	0.890	0.897	0.905	0.878
EFFICIENCY	(T/S)	0.873	0.866	0.689	0.703
PRESSURE RATIO	(T/T)	1.64	1.76	1.18	1.19
PRESSURE RATIO	(T/S)	1.66	1.79	1.24	1.25
POWER	(HP)	15971.	16602.	4540.	4617.
SPEED	(RPM)	10953.	10953.	5014.	5000.
DH ACT	(BTU/LB)	45.1	46.9	12.8	13.0
MEAN DIAMETER	(IN)	17.30	17.65	12.60	12.90
BLADE HEIGHT	(IN)	0.700	1.050	1.600	1.900
VEL.RATIO, ACTUAL		0.550	0.551	0.344	0.349
MAX TIP SPEED	(FT/SEC)	861.	894.	311.	324.
AN**2 X 10**8		46.	70.	16.	19.
EFFECT. AREA	(IN**2)	4.830	7.535	17.602	19.756
GAS CON.	(FT-LB/LB-R)	96.3	96.3	96.3	96.3
GAMMA		1.332	1.332	1.358	1.358
EXIT MACH NUMBER		0.122	0.181	0.279	0.259

PUMP PERFORMANCE CHARACTERISTICS

		MAIN FUEL PUMP STAGE 1	MAIN FUEL PUMP STAGE 2	MAIN OXID. PUMP STAGE 1
EFFICIENCY		0.694	0.565	0.796
POWER	(HP)	17119.	15454.	9157.
SPEED	(RPM)	10953.	10953.	5014.
NPSH AVAILABLE	(FT)	177.7	13655.8	62.6
SS SPEED		25063.	742.	25158.
S SPEED		787.	494.	1519.
HEAD RISE	(FT)	13580.	17402.	2379.
DIAMETER , EXIT	(IN)	17.95	20.51	18.22
TIP SPEED	(FT/SEC)	859.	981.	399.
VOL FLOW , EXIT	(GPM)	8139.	4703.	10605.
HEAD COEF.,		0.593	0.568	0.481
FLOW COEF.,		0.103	0.113	0.140

Table 4.2.1.6-4. Derivative STBE Split Expander Costs

Total Development Cost (DDT&E), M$500*
Production Cost (TFU), M$8.1
Operations Cost/Launch/Engine, M$0.126**
Constant FY87$

*Applies to Derivative STBE Split Expander; an additional M$900 Development Program is estimated for the STME Split Expander.
**Based on the 100th mission, 10 missions per year, and seven boosters engines.

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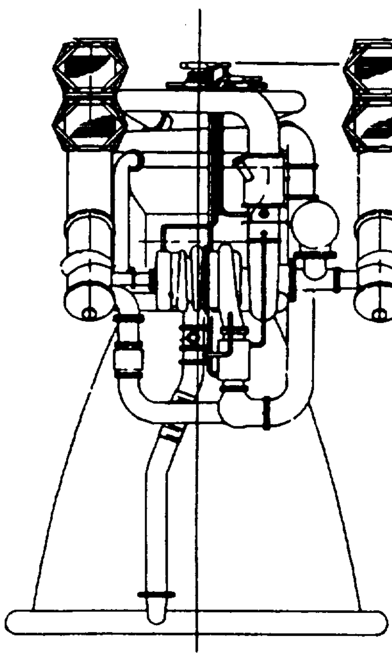


Propellants	CH_4/LO_2
Mixture Ratio	3.5
Chamber Pressure - psia	877
Thrust - Vacuum	762,900
Sea Level - lb	625,000
Specific Impulse - Vacuum	342.8
Sea Level - sec	280.8
Nozzle Area Ratio	20
Diameter - in.	136
Length - in.	205
Weight - lb	6394

FD 357542

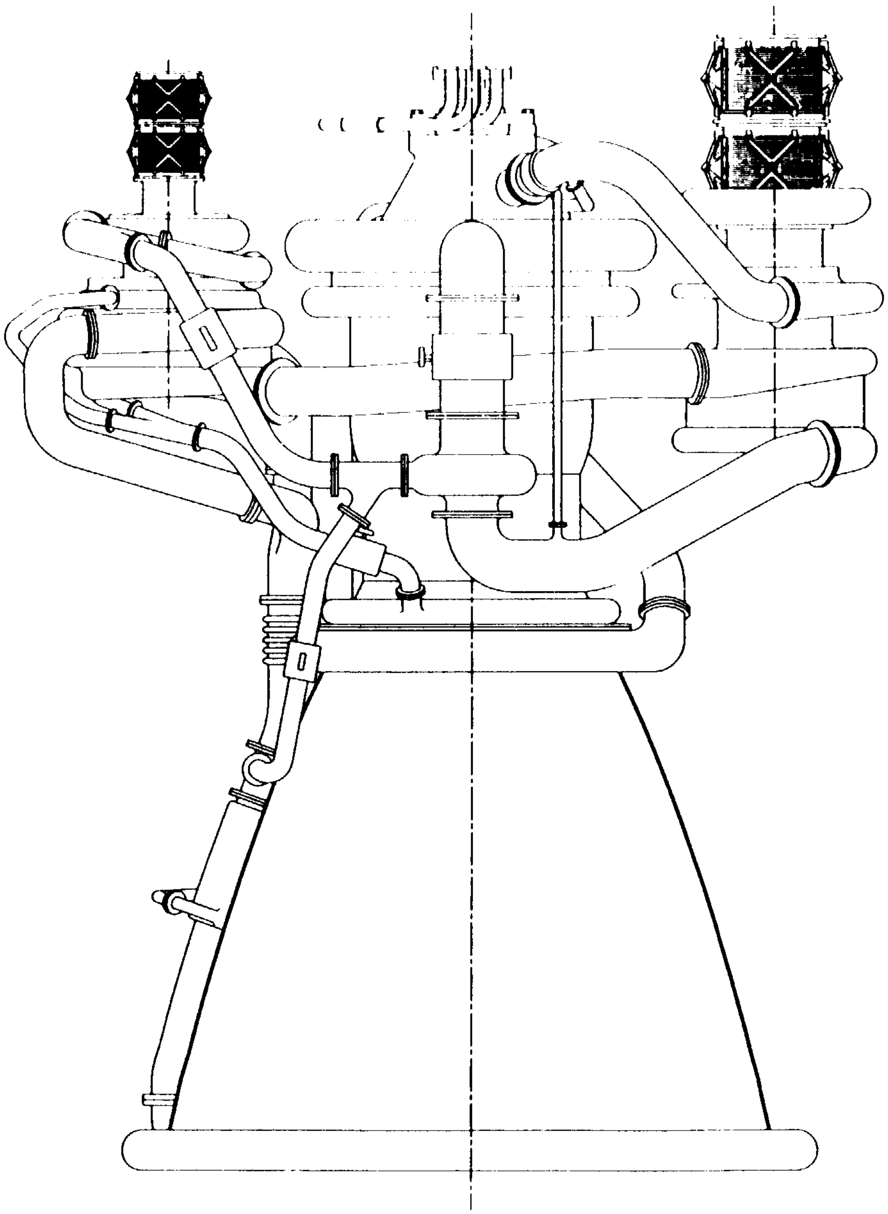
Figure 4.2.2.1-1. STBE LO_2/CH_4Unique Split Expander Engine at Normal Operating Conditions

4.2.2.2 Engine Cycle

The STBE (SE) is a split expander cycle with liquid oxygen and liquid methane as the propellants. This engine operates at a main chamber pressure of 764.5 psia at the normal power level (NPL) of 625K lb and has the capability of running at a design power level of 750K lb. The nozzle area ratio is optimized, for a booster engine application, at 13.5:1 and results in a delivered sea level specific impulse of 281.4 seconds at NPL. Figure 4.2.2.1-2 presents selected engine characteristics at the normal power level.

4.2.2.2.1 Flowpath Description

A simplified flow schematic for the STBE (SE) is presented in Figure 4.2.2.2-1 showing only the major flow paths and components.



	Normal Power Level	Design Power Level
Propellants	LO_2/CH_4	LO_2/CH_4
Mixture Ratio	3.5	3.5
Chamber Pressure	765 psia	895 psia
Thrust - Vacuum	730,389 lb	855,390 lb
- Sea Level	625,000 lb	750,000 lb
Specific Impulse - Vacuum	328.9 sec	329.3 sec
- Sea Level	281.4 sec	288.7 sec
Nozzle Area Ratio	13.5	13.5
Diameter	109 in.	109 in.
Length	170 in.	170 in.
Weight	TBD lb	TBD lb

FD 359990

Figure 4.2.2.1-2. STBE LO_2/CH_4 Unique Split Expander Engine at Design Operating Conditions

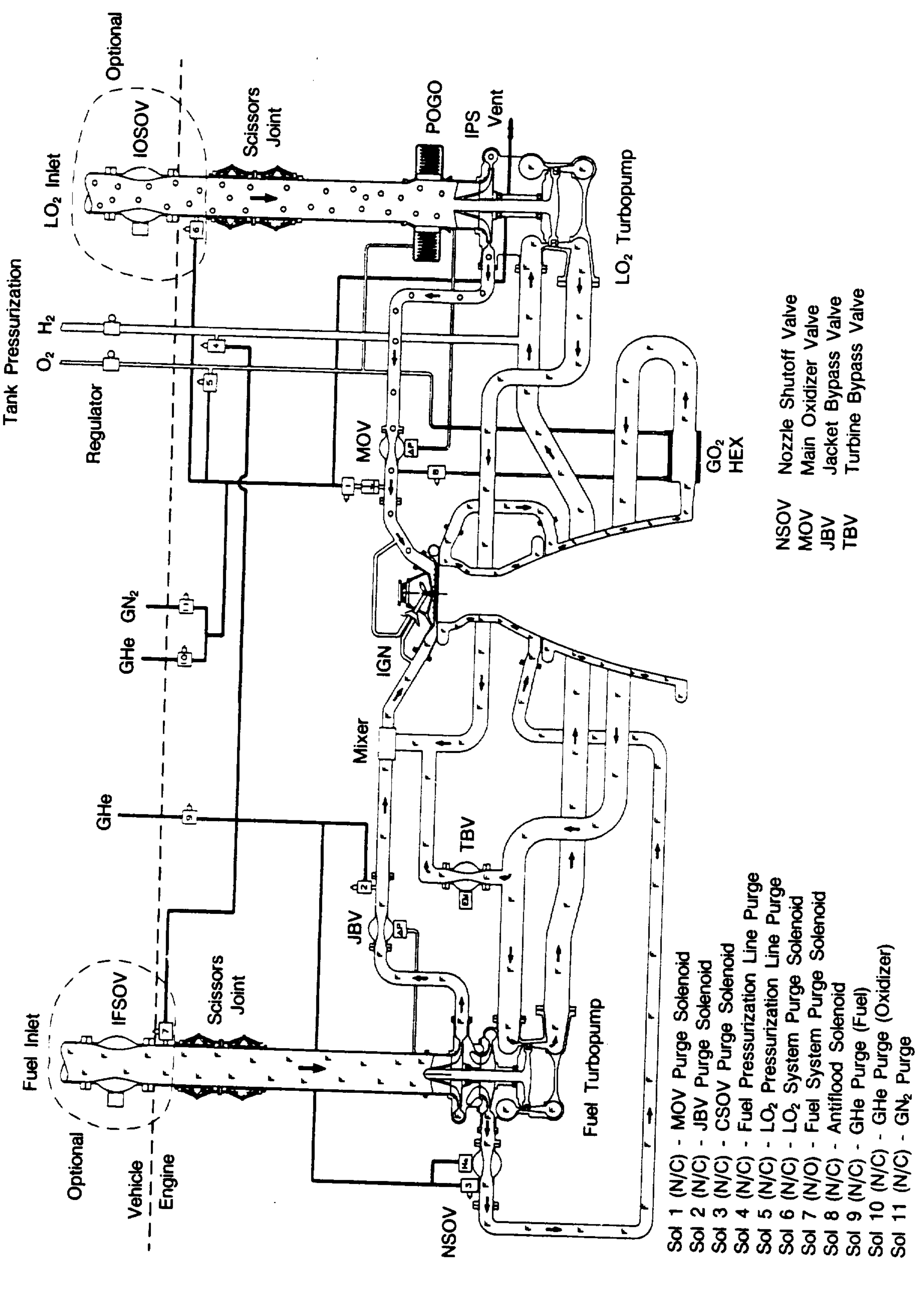


Figure 4.2.2.2-1. Simplified Flow Schematic for STBE LO_2/CH_4 Unique Split Expander Engine

Liquid oxygen and methane enter the engine at a NPSH level, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps. No boost pumps are required in these systems.

At normal power level, the methane pump operates at 9,689 rpm to provide a first stage pump discharge pressure level of 1098.6 psia. From the first stage pump exit, 44 percent of the fuel is sent through a control valve (JBV) to a mixer downstream of the turbines bypassing the chamber jacket and turbines. The remaining 56 percent of the flow is routed to the second stage of the methane pump. The second-stage pump discharge level is 4072 psia. From the second stage pump exit, the methane is routed through the nozzle shutoff valve into the chamber wall passages where there is counterflow cooling and then through the tubular nozzle wall where there is parallel flow cooling. This heated methane is then used to provide power to drive the propellant pumps. 87.3 percent of the nozzle cooling flow is routed through the turbines. The hot (920 R) methane gas is initially expanded through the methane pump drive turbine and is subsequently routed to a second turbine that powers the oxygen pump. The turbine exhaust is then routed to a mixer, where it combines with the remainder of the methane flow, and is then injected into the main chamber.

At normal power level, the oxidizer operates at 4054 rpm to provide an oxygen pressure level of 978.0 psia. From the pump exit, the oxygen flow is routed through a control valve and injected directly into the main chamber.

Some key design conditions for the pumps are listed in Table 4.2.2.2-1.

Table 4.2.2.2-1. *Unique STBE Split-Expander Design Conditions*

HPFTP	
Pressure — psia	4072.
Speed — rpm	9689.
Turbine Temperature — R	920.
Pump Stages	2
Turbine Stages	1
HPOTP	
Pressure — psia	978.
Speed — rpm	4054.
Turbine Temperature — R	826.
Pump Stages	1
Turbine Stages	1
Main Chamber	
Chamber Pressure — psia	765.
Heat Pickup — btu/sec	154,994
Coolant Flow — lbm/sec	275.

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4.2.2.2.2 *Engine Operation*

The engine is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor or on the pad, because all fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the LO_2 injector is primed prior to opening the fuel shutoff valve to assure liquid oxidizer flow. During the start and shutdown, a small helium purge is used in the main chamber injector to eliminate the danger of hot gas flow reversals during transient

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operation. Main chamber ignition will be accomplished with an electrical spark-excited, oxygen/methane torch igniter (regeneratively cooled mini-rocket).

Engine operation is controlled by a timed sequence of the five valves (NSOV, JBV, FSOV, TBV, MOV in Figure 4.2.2.2-1). Engine acceleration is accomplished by scheduling the valves on open-loop schedules to full thrust.

During preconditioning, all of the valves are closed except for the main oxidizer valve (MOV), which is approximately 25 percent open for simultaneous LO_2 injector cooldown. Once the engine is adequately preconditioned, the MOV opens further to completely fill the LO_2 injector prior to ignition. During the process of filling the LO_2 injector, the nozzle shutoff valve (NSOV) remains closed to prevent a cooling of the nozzle/chamber cooling jacket. Once the LO_2 injector is full, the NSOV and the fuel shutoff valve (FSOV) are opened so the fuel can flow freely to the injector. At this point, the jacket bypass valve (JBV) and the turbine bypass valve (TBV) remain closed so as to force all of the available fuel through the turbines. After ignition and upon sufficient power from the turbines, the jacket bypass valve (JBV) opens to bypass flow from the pump first-stage discharge to the mixer. Once the desired thrust level is achieved, the TBV opens to control turbine power. At this point, the engine should be at its steady-state conditions.

Engine shutdown is accomplished using a time based scheduling of the propellant valves. First, the turbine bypass valve (TBV) is further opened to reduce turbine power and slow the pumps. Then the methane system is shut down by closing the jacket bypass valve (JBV), nozzle shut off valve (NSOV) and fuel shut off valve (FSOV), in that order, to purge the fuel system of excess methane. Finally, the oxidizer system is shut down by closing the main oxidizer valve (MOV).

4.2.2.3 Turbomachinery

4.2.2.3.1 Unique STBE Split Expander Oxidizer Turbopump

The oxidizer turbopump, shown in Figure 4.2.2.3-1, is configured as a single centrifugal shrouded impeller pump with an inlet inducer driven by a two-stage axial flow turbine. The inducer and impeller, made of fine grained cast and HIP Inconel 718, are coupled to the turbine through a single turbine disk with an integral shaft made of Waspaloy. Pump and turbine inlet and discharge housings are fabricated from aluminum to minimize machining costs. The turbine, being an integrally bladed rotor, has its blades formed by EDM process. One ball and roller bearing, made of 440C material will be used to support the pump rotor system.

The mechanical description of the features of this turbopump are the same as the STBE Derivative Gas Generator Oxidizer turbopump.

4.2.2.3.2 Unique STBE Split Expander Fuel Turbopump

The fuel turbopump is configured as an inlet inducer with a two-stage centrifugal shrouded impeller pump driven by a single-stage axial flow turbine. The inducer, made of aluminum, and the impellers, made of titanium A110 ELI, are coupled to the turbine through a single turbine disk with an integral shaft made of Waspaloy. Pump and turbine inlet and discharge housings are fabricated from aluminum to minimize machining costs. The turbine, being an integrally bladed rotor, has its blades formed by EDM process. One ball and roller bearing, made of 440C material will be used to support the pump rotor system. Figure 4.2.2.3-2 shows the fuel turbopump and its major components.

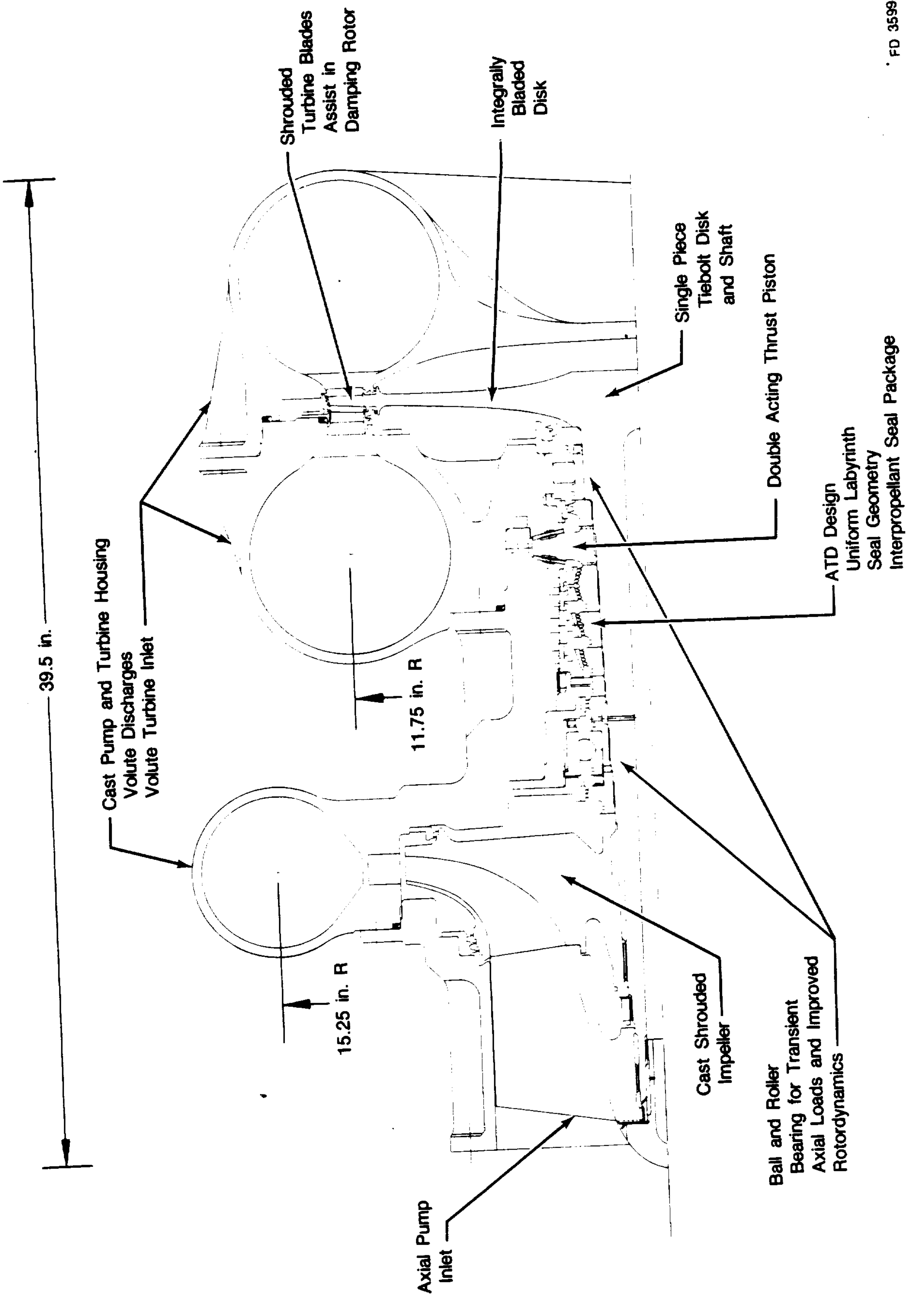

Figure 4.2.2.3-1. STBE Unique Split Expander Oxidizer Turbopump

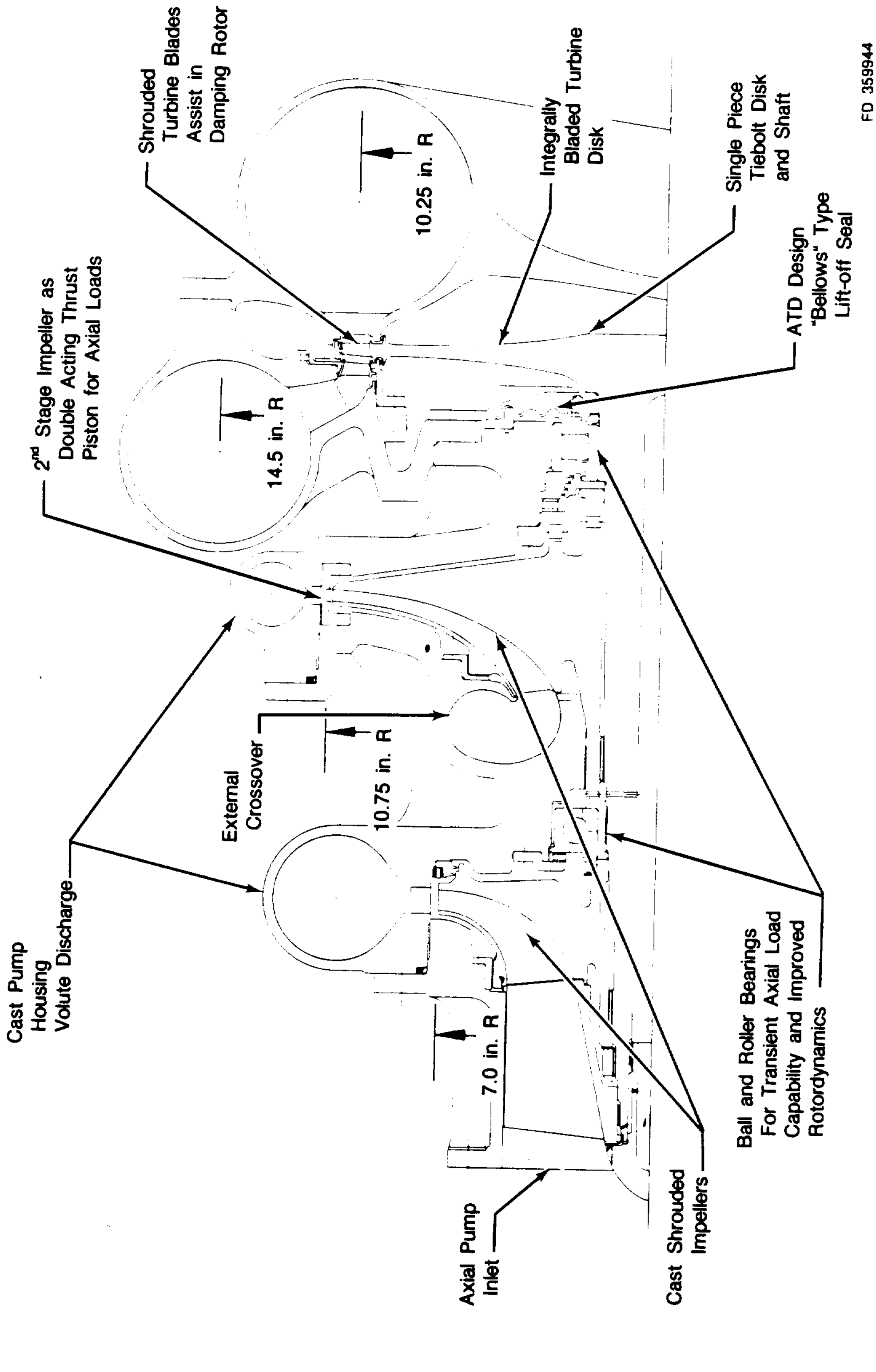

Figure 4.2.2.3-2. STBE Unique Split Expander Fuel Turbopump

The mechanical description of the features of this turbopump are the same as the STBE Derivative Gas Generator Fuel turbopump.

4.2.2.4 Combustor

The unique STBE split expander engine minimum chamber volume, injector design, and acoustic liner design were determined using the procedures outlined in section 4.1.1.4 for the derived STBE engine. The unique STBE chamber and injector element designs are summarized in Table 4.2.2.4-1. An L* of 49 inches is required to meet the specified 98 percent characteristic velocity efficiency. This is greater than that of the gas generator cycle engines mainly as a result of poorer atomization in the split expander due to less available pressure drop across the injector elements.

Table 4.2.2.4-1. *Unique STBE Split Expander Combustor and Injector Design*

Chamber L* (Min)-in.	49
Fuel Flow-lb/sec	577.2
ΔP Fuel-psi	75.4
LO_2 Flow-lb/sec	2020.1
ΔP LO_2 psi	58.1
No. of Elements	1632
Spud ID-in.	0.303
Annular Gap-in.	0.03

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The acoustic liner design of the unique STBE chamber is given in Table 4.2.2.4-2. The acoustic absorption of the liner is 20.6 percent at the first tangential frequency (750 Hz). This relatively low absorption is primarily a result of the low resonant frequency and high Mach numbers that exist in the chamber due to its diameter (38.11 in.) and contraction ratio (2.0).

Table 4.2.2.4-2. *Acoustic Liner Design*

Chamber Pressure-psi	895.3
Aperture — Gas Temperature-°R	2000
Aperture — Gas Molecular Wt.	21.8
Hole Diameter-in.	0.10
Hole Length-in.	0.35
Area Ratio	0.05
Backing Cavity Depth-in.	0.60
Liner Length-in.	7.24

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4.2.2.4.1 Main Injector

The mechanical description of the features of this main injector are the same as the STBE Derivative Gas Generator main injector, with the exception that this injector has a toroidal fuel inlet manifold. The main injector assembly, injector element, and injector pattern are shown in Figures 4.2.2.4-1 through -3, respectively.

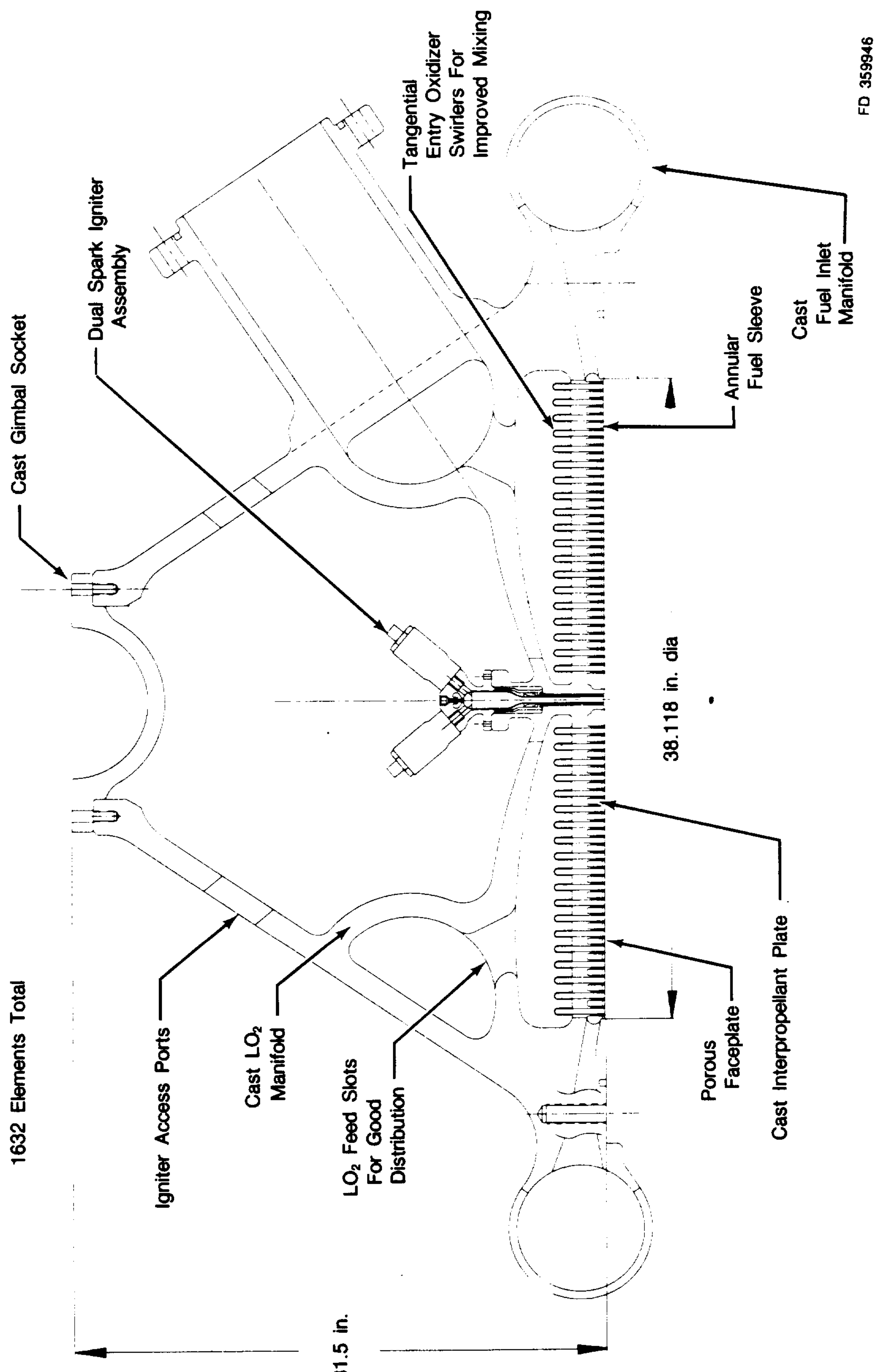

Figure 4.2.2.4-1. STBE Unique Split Expander Main Injector

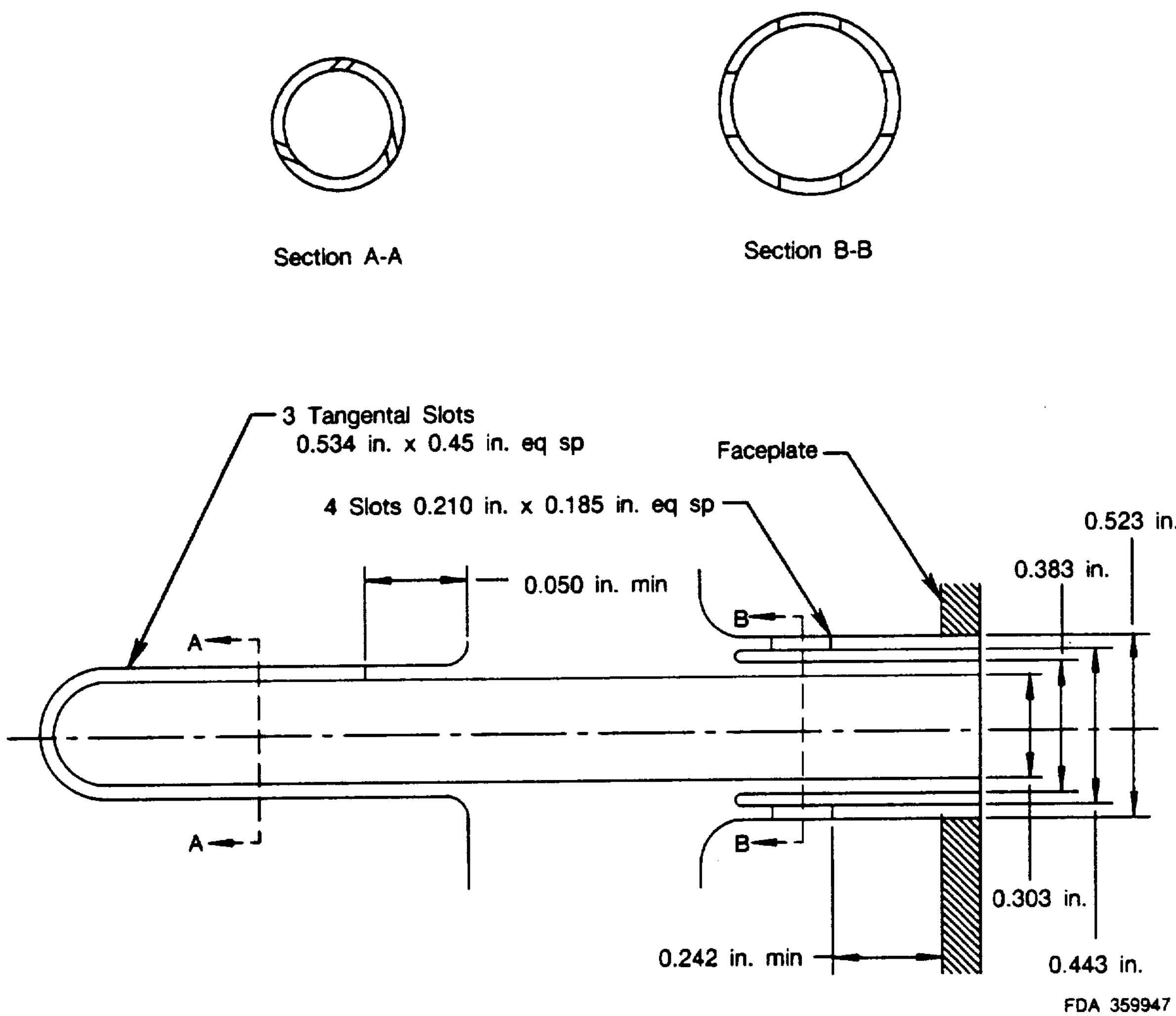

Figure 4.2.2.4-2. STBE Unique Split Expander Main Injector Element

4.2.2.4.2 Combustion Chamber

The combustion chamber is regeneratively cooled by fuel from the high pressure pump discharge. The fuel enters the tubular cooling jacket through the inlet manifold below the throat. The coolant then flows forward, counter to the gas path flow, to the throat. The fuel then cools the chamber wall, is collected, and exits through the toroidal manifold. This flow configuration provides the coolest fuel at the throat where wall heat flux is highest. The combustion chamber assembly and its major features are shown in Figure 4.2.2.4-4.

The main combustion chamber is constructed of 630 double-taper Haynes 230 nickel alloy tubes brazed together. Coolant inlet and exit manifolds are brazed to each end of the chamber. Simultaneously, a structural jacket is brazed to the backside of the tubes to contain the hoop loads due to the combustion chamber pressure. The axial thrust load from the nozzle is carried by the structural jacket and an exterior cylindrical collar welded to each manifold.

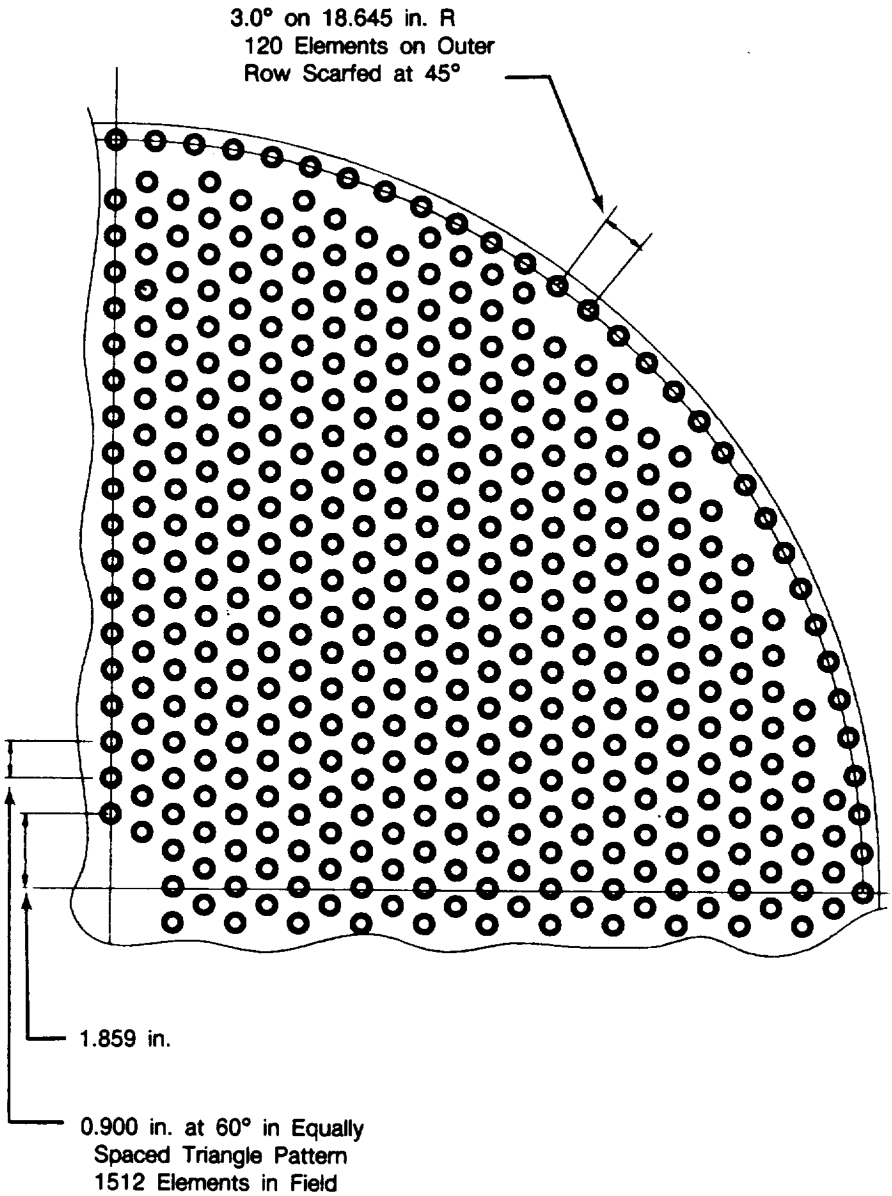

Figure 4.2.2.4-3. STBE Unique Split Expander Main Injector Pattern

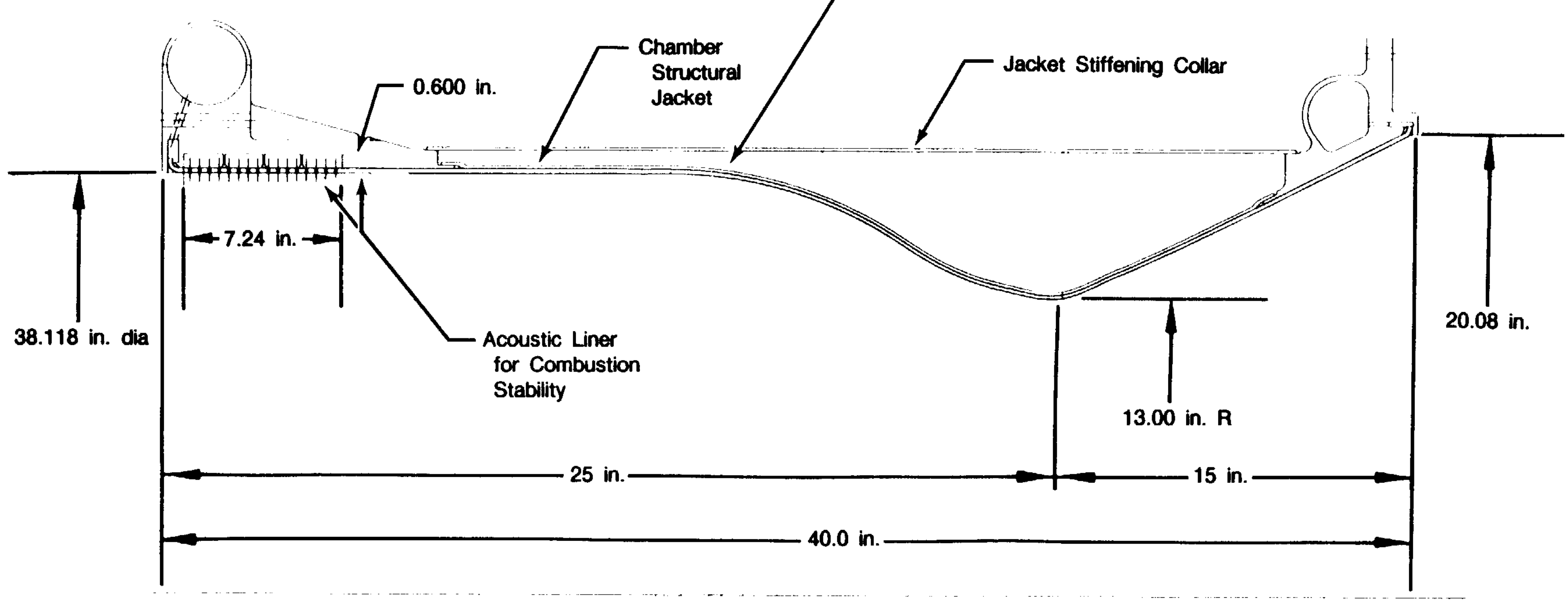

Figure 4.2.2.4-4. STBE Unique Split Expander Main Combustion Chamber

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An acoustic cavity is positioned adjacent to the injector face to provide combustion stability. The integral cavity is machined into the coolant exit manifold structure. This cavity is connected to the combustion chamber cavity through a specified number and size of holes formed by pressing small tubes between the Haynes 230 tubes, locally deforming the tubes prior to brazing. Figure 4.2.2.4-5 shows the acoustic liner hole pattern through the chamber liner. After brazing, the small tube ID forms the communicating cavity between the combustion chamber and the acoustic cavity. A liner is placed in the acoustic cavity to which a minimal amount of coolant flow is tapped off the chamber coolant exit and used to cool the backside of the acoustic cavity. This coolant is then dumped into the cavity to provide a purged outflow, preventing hot gas ingestion into the acoustic cavity.

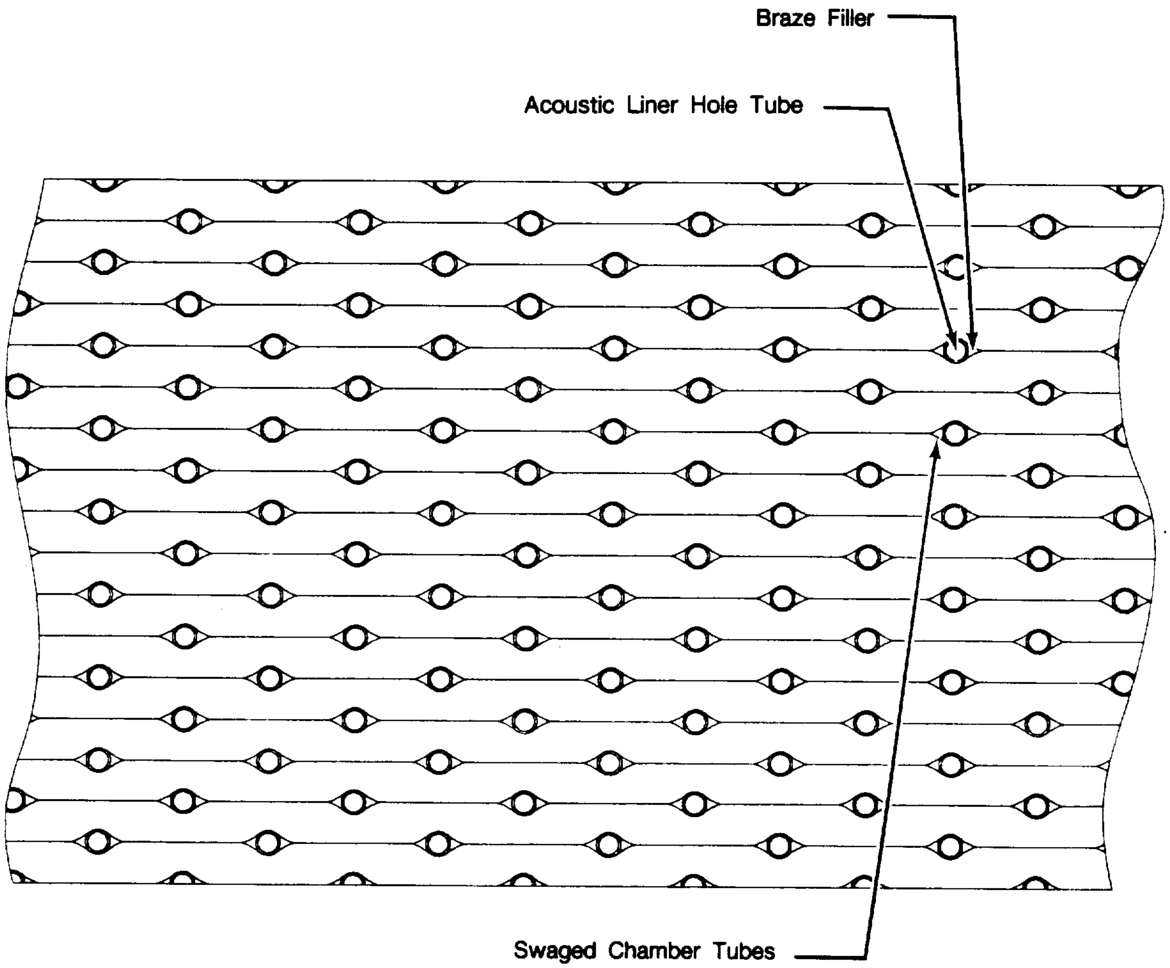


Figure 4.2.2.4-5. STBE Unique Split Expander Main Combustion Chamber Acoustic Liner Hole Pattern Through the MCC Liner

The thrust chamber design uses a brazed assembly of 720 double tapered, constant wall thickness Haynes 230 tubes. The chamber extends to a nozzle expansion area ratio of 2.41:1, has an injector diameter of 36.77 inches and a throat diameter of 26.00 inches, with a corresponding contraction ratio of 20:1. Acoustic apertures are located within the braze joints at the forward end of the chamber. A counterflow cooling system that uses 53 percent of the methane fuel flow is used to regeneratively cool the nozzle. The coolant tube dimensions are sized to meet the heat

transfer and cycle requirements at the 750K lbf sea level thrust at 896 psia chamber pressure design point and reflect the following design guidelines.

- Maximum stress < 0.2 percent yield strength.
- Ultimate tube temperature margin > 375 R.
- Coolant Mach number < 0.5.
- Cooling enhancement from tube curvature.

Figure 4.2.2.4-6 summarizes the throat chamber contour and tube geometry.

The methane coolant enters the tube assembly at 241 R and 5315 psia and exits at 626 R and 3975 psia. The maximum predicted values of hot wall temperature and heat flux are 2170 R and 21 Btu/in.^{2}-sec, respectively. The highest calculated coolant Mach number is 0.2. Figure 4.2.2.4-7 summarizes the predicted thermal performance characteristics for the thrust chamber.

4.2.2.4.3 *Torch Igniter*

A continuous burning torch igniter was chosen for use in the main combustion system because of the simplicity of the design and reliability in tests. The igniter configuration employed evolved from development efforts since 1957 at Pratt & Whitney and is based on experience gained from the successful RL10 and XLR-129 engine programs.

In the main combustion chamber, the torch is mounted axially in the center of the injector, directing the torch down along the centerline of the combustion chamber.

The construction of the torch assembly is discussed in Space Transportation Main Engine Configuration Study, P&W FR-19830-1 Volume II, page 93.

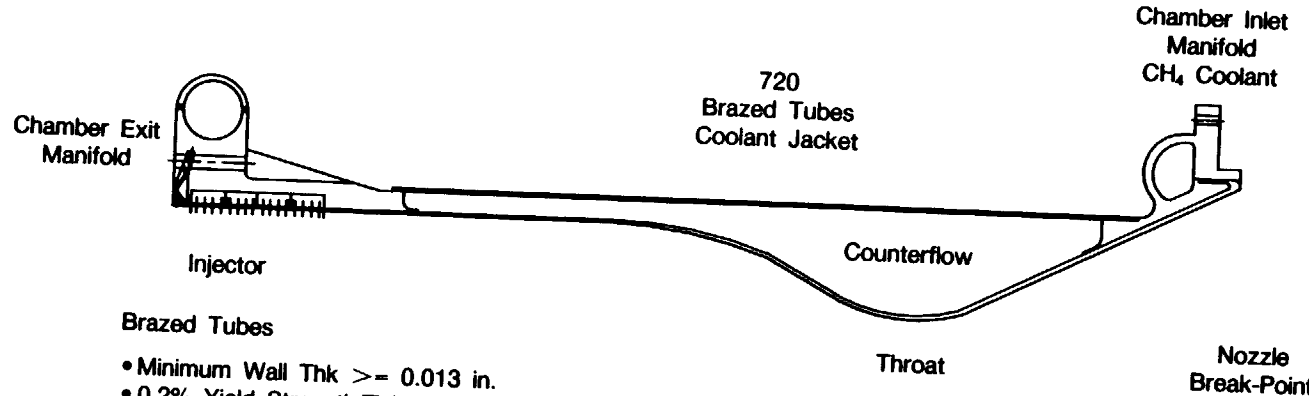
4.2.2.4.4 *Regeneratively Cooled Nozzle*

The regeneratively cooled nozzle design uses a brazed assembly of 1170 single tapered, constant wall thickness Haynes 230 tubes, is 80-inches long and extends from an expansion area ratio of 2.41:1 to an exit area ratio of 13.5:1. Figure 4.2.2.4-8 shows the regeneratively cooled nozzle with its overall dimensions. A parallel flow cooling system using 53 percent of the methane fuel flow (309 lb/sec) is used. The methane is used to cool the thrust chamber prior to cooling the nozzle. The nozzle tube dimensions are sized to meet the heat transfer and cycle requirements at the 750K lbf sea level thrust at 896 psia chamber pressure design point. The tube geometry reflects the following design guidelines.

- Maximum stress < 0.2 percent yield strength.
- Ultimate tube temperature margin > 375R.
- Coolant Mach number < 0.5.

Figure 4.2.2.4-9 summarizes the nozzle contour and tube geometry.

The methane coolant enters the nozzle tube assembly at 626 R and 3875 psia and exits at 930 R and 3168 psia. The maximum predicted values of hot wall temperature and heat flux are 1425 R and 9.2 Btu/in.^{2}-sec, respectively. The highest calculated coolant Mach number is 0.21. Figure 4.2.2.4-10 summarizes the predicted thermal performance characteristics for the regeneratively cooled nozzle.



Brazed Tubes

- Minimum Wall Thk >= 0.013 in.
- 0.2% Yield Strength/Tube Stress >= 1.0
- Coolant Mach No. <= 0.5
- Maximum Wall Temperature Limit <= 2260 R
- Minimum UTTM >= 375 R

Thrust Chamber Contour Data

- Chamber Length = 40 inches
- Divergent Nozzle Length = 15 inches
- Throat Diameter = 26.00 inches
- Injector Diameter = 36.77 inches
- Contraction Ratio = 2.0
- Divergent Nozzle Area Ratio = 2.41
- L_* = 71.3 inches
- η_{c*} (Throat) = 0.99 $^+$
- Number of Tubes = 720
- Liner Construction - Brazed Tubes
- Liner Material - Haynes 230

Thrust Chamber Cooling Tube Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)	Aspect Ratio
15.00	20.19	0.175	0.193	0.013	1.105
9.95	17.66	0.153	0.169	0.013	1.103
1.88	13.86	0.119	0.155	0.013	1.300
0.87	13.40	0.115	0.157	0.013	1.365
-0.14	13.00	0.112	0.159	0.013	1.423
-2.66	13.28	0.114	0.158	0.013	1.381
-7.20	15.18	0.131	0.149	0.013	1.134
-10.73	17.06	0.148	0.163	0.013	1.101
-20.32	18.39	0.159	0.176	0.013	1.103
-30.41	18.39	0.159	0.176	0.013	1.103
-40.00	18.39	0.159	0.176	0.013	1.103

Figure 4.2.2.4-6. STBE Unique Split Expander Thrust Chamber Cooling Design Configuration

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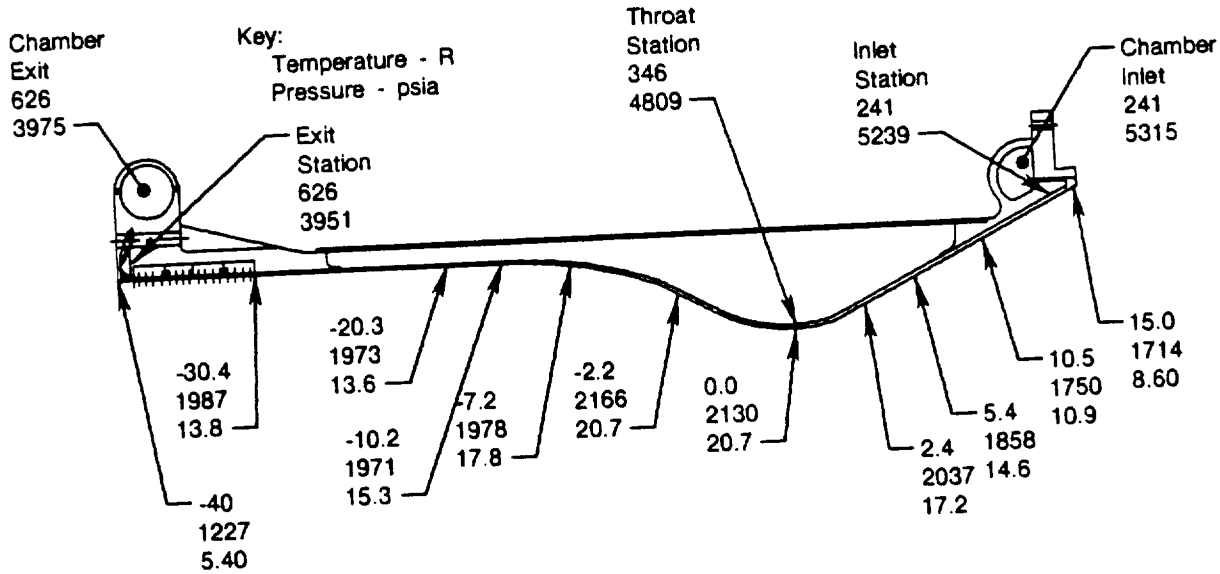


Coolant Performance
Thrust = 120%
M_{cool} = 309 lbm/sec

Thrust Chamber Heat Transfer Performance

SL Thrust - lbf	750K
Chamber Pressure - psia	896
Propellant	LO_2/CH_4
O/F Ratio	3.50
Coolant Flow - lbm/sec	309
Inlet Temperature - R	241
Exit Temperature - R	626
Coolant Heat Pickup - Btu/sec	97,829
Inlet Pressure - psia	5,315
Exit Pressure - psia	3,975
Pressure Drop - psid	1,340

Hot Wall Temperature and Heat Flux

Key:

Axial Location - in.
Wall Temp - R
Heat Flux - Btu/in.^{2} - sec

FDA 363349

Figure 4.2.2.4-7. STBE Unique Split Expander Thrust Chamber Heat Transfer Performance Summary

4.2.2.5 Controls

The description of the engine controls for the Unique Split Expander Cycle Engine is similar to that for the Derivative STBE Split Expander Engine.

4.2.2.6 Engine Configuration and Integration

4.2.2.6.1 Unique STBE Split Expander Engine Assembly

The arrangement of the external configuration of the engine was based on considerations of accessibility for routine component inspections, removals and replacements. Figures 4.2.2.6-1 and -2 show the side and top views of the engine assembly and its major components.

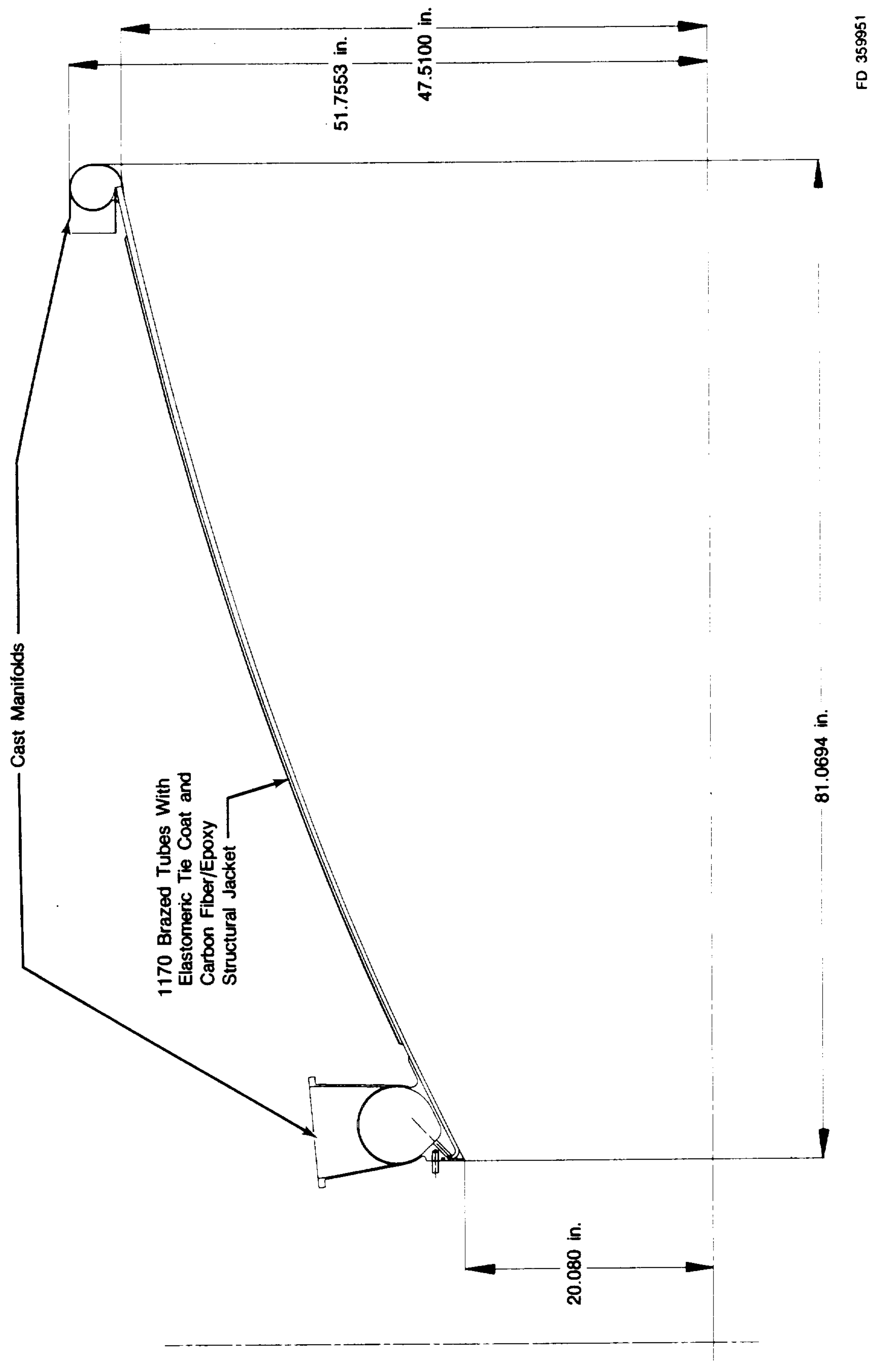
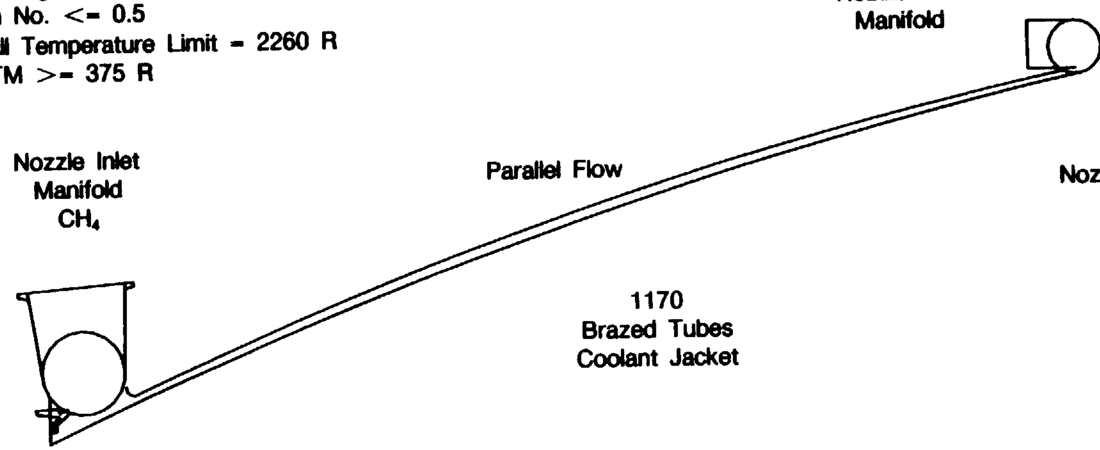


Figure 4.2.2.4-8. STBE Unique Split Expander Nozzle

Brazed Tubes

Minimum Wall Thickness >= 0.013 in.
0.2% Yield Strength/ Tube Stress = 1.0
Coolant Mach No. <= 0.5
Maximum Wall Temperature Limit = 2260 R
Minimum UTTM >= 375 R



Nozzle Contour Data

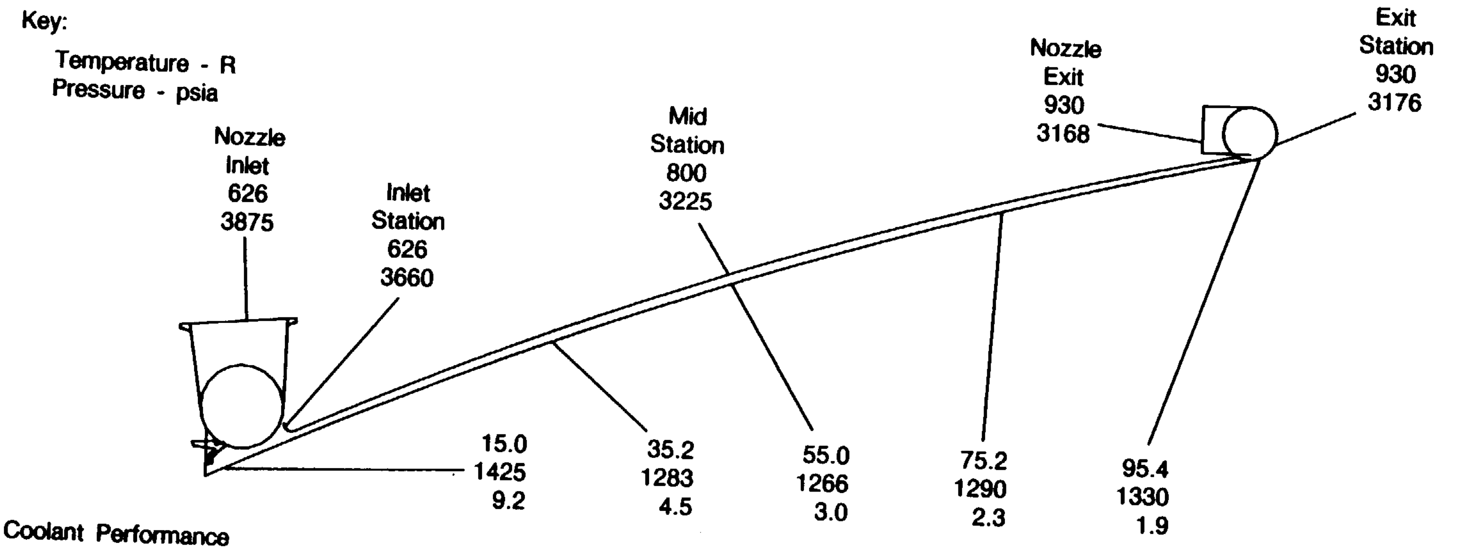
Nozzle Length = 80 inches
Inlet Expansion Ratio = 2.41
Exit Expansion Ratio = 13.5
Number of Tubes = 1170
Nozzle Construction - Brazed Tubes
Nozzle Material - Haynes 230

Nozzle Cooling Tube Geometry

Axial Length (in.)	Wall Radius (in.)	OD Width (in.)	OD Height (in.)	Wall Thickness (in.)	Aspect Ratio
15.00	20.19	0.107	0.180	0.013	1.688
25.11	25.08	0.133	0.165	0.013	1.240
36.74	30.11	0.160	0.177	0.013	1.103
45.84	33.59	0.179	0.198	0.013	1.107
54.95	36.80	0.196	0.217	0.013	1.108
64.05	39.70	0.212	0.235	0.013	1.109
76.18	43.20	0.231	0.256	0.013	1.110
85.28	45.54	0.243	0.270	0.013	1.111
95.40	47.88	0.256	0.285	0.013	1.114

FDA 363807

Figure 4.2.2.4-9. STBE Unique Split Expander Nozzle Cooling Design Configuration



Coolant Performance

Thrust = 120%
M_{cool} = 309 lbm/ sec

Nozzle Heat Transfer Performance

SL Thrust - lbf	750K
Chamber Pressure - psia	896
Propellant	LO_2/ CH_4
O/F Ratio	3.50
Coolant Flow - lbm/ sec	309
Inlet Temperature - R	626
Exit Temperature - R	930
Coolant Heat Pickup - Btu/ sec	73,117
Inlet Pressure - psia	3875
Exit Pressure - psia	3168
Pressure Drop - psid	707

Hot Wall Temperature and Heat Flux

Key:

Axial Location - in.
Wall Temp - R
Heat Flux - Btu/ in.^{2} - sec

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Figure 4.2.2.4-10. STBE Unique Split Expander Nozzle Heat Transfer Performance Summary

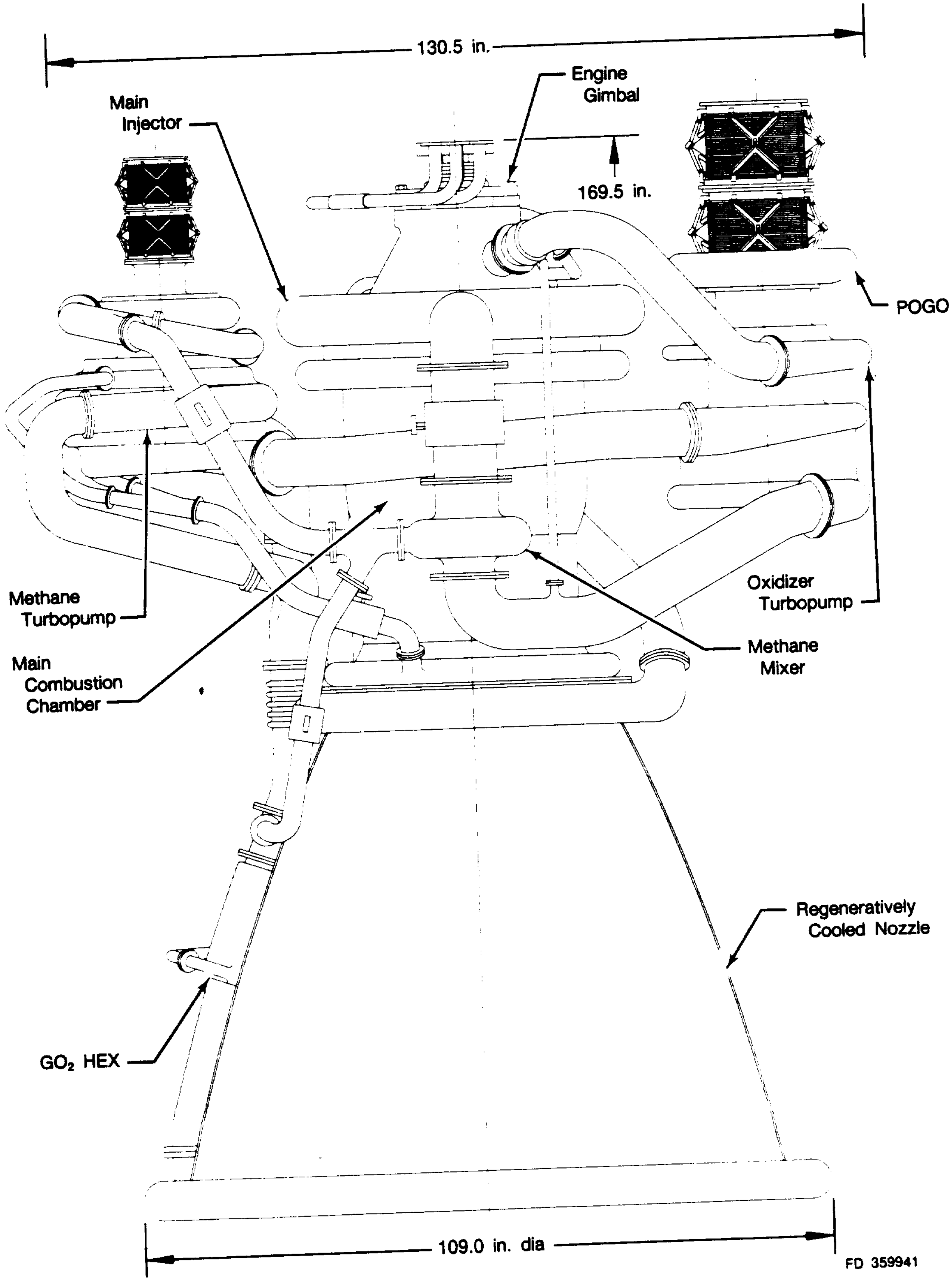

Figure 4.2.2.6-1. STBE Unique Split Expander Engine Assembly — Side View

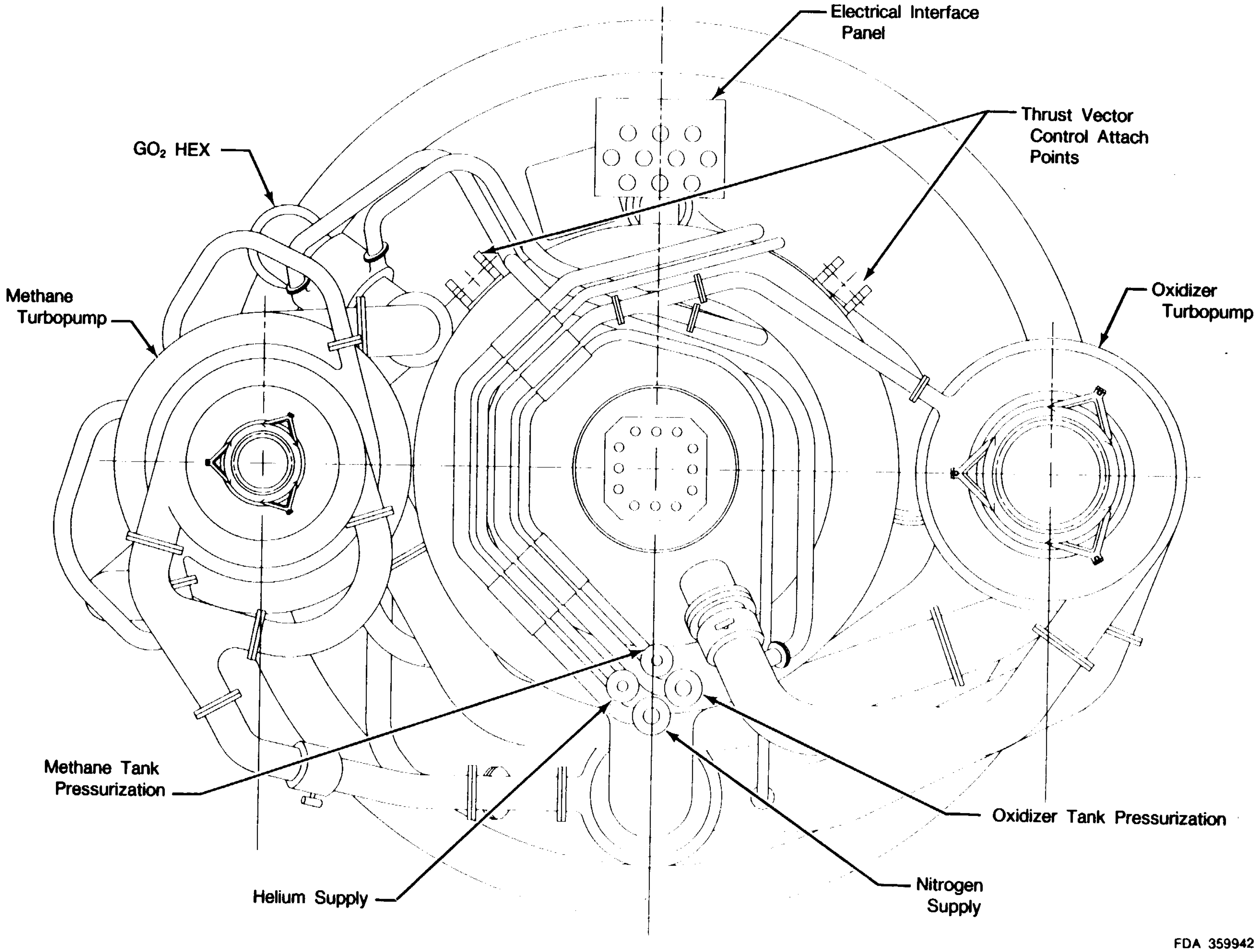

Figure 4.2.2.6-2. STBE Unique Split Expander Engine Assembly — Top View

4.2.2.6.2 *Split Expander GO_2 Heat Exchanger*

The GO_2 heat exchanger provides gaseous oxygen to the oxygen tank for tank pressurization. The GO_2 heat exchanger uses oxidizer turbine exhaust duct flow as the heat source to vaporize the liquid oxygen as shown in Figure 4.2.2.2-1. The heat exchanger consists of five Haynes 214 stainless steel tubes wrapped in parallel around the exhaust duct. The exhaust duct wall is made of beryllium copper with trip-strip roughened walls to enhance the heat transfer. The tubes are packed in powdered copper to structurally isolate the tubes from the duct wall, while providing a good heat transfer medium. This design eliminates the possibility of accidental mixing of the oxygen and exhaust turbine flow, thereby eliminating a category 1 failure mode.

The GO_2 heat exchanger requires five 3/8-inch diameter tubes 42.5-feet long, wrapped around the 7-inch duct. The tubes have 0.015-inch thick walls and are separated from one another by 0.050-inch, requiring a total duct length of 3.90 feet. Figure 4.2.2.6-3 diagramatically presents the GO_2 heat exchanger geometry. The GO_2 heat exchanger has been thermally analyzed for the STBE 120 percent engine operating point with an oxygen flow rate of 10.1 lbm/sec. The heat exchanger is designed to supply 400 R oxygen to the tank. Figure 4.2.2.6-3 also summarizes the predicted heat exchanger performance.

4.2.2.6.3 *Engine Performance*

The STBE (SE) system performance was determined during the preliminary design using the accepted JANNAF methodology. Table 4.2.2.6-1 lists the detailed analysis for the design power level (DPL) of 750K lbf sea level thrust while the normal power level (NPL) of 625K lbf sea level thrust is given in Table 4.2.2.6-2.

During this study program, detailed aerothermal analyses were made to predict component performance levels and these were incorporated into a steady state model of the complete engine. Simplified flow schematics are presented in Figures 4.2.2.6-4 and -5 with key operating parameters noted for each thrust level. Tables 4.2.2.6-3 and -4 define performance of the individual components and their operating environments for the STBE (SE) at NPL (100 percent) and at the design power level (120 percent) respectively.

The STBE (SE) uses an external GO_2 heat exchanger to pressurize the LO_2 tank. This eliminates a category 1 failure mode. The heat exchanger uses the hot methane jacket exit flow to vaporize a small amount of LO_2 which is returned to the tank. The methane tank is pressurized with gaseous methane tapped off at the exit of the second turbine that drives the oxidizer main pump.

4.2.2.6.4 *Engine Costs*

This section summarizes cost estimates for the 750K SL thrust, 895 psia chamber pressure, Unique STBE Split Expander cycle. Table 4.2.2.6-5 summarizes significant costs for the engine.

The DDT&E Cost includes all of the functions required to design, develop, test and evaluate the engine system. All of the DDT&E functions shown in the ALS engine WBS (see Volume III) have been included. Development Cost is based on a 90-month phase C/D program with 960 engine firings for the STME Unique Split Expander and 488 for the Unique STBE Split Expander. Sufficient accountable firings have been included in the program to demonstrate 0.99 engine reliability with one failure.

STBE GO_2 Heat Exchanger
Thermal and Flow Performance

Coolant Exit Duct

Thrust - %	120
Inlet Temperature - R	952
Exit Temperature - R	947
Inlet Pressure - psia	3168
Exit Pressure - psia	3165
Flow Rate - lbm/sec	309

Oxygen

Flow Rate - lbm/sec	10.1
Inlet Temperature - R	169
Exit Temperature - R	400
Inlet Pressure - psia	1223
Exit Pressure - psia	1006

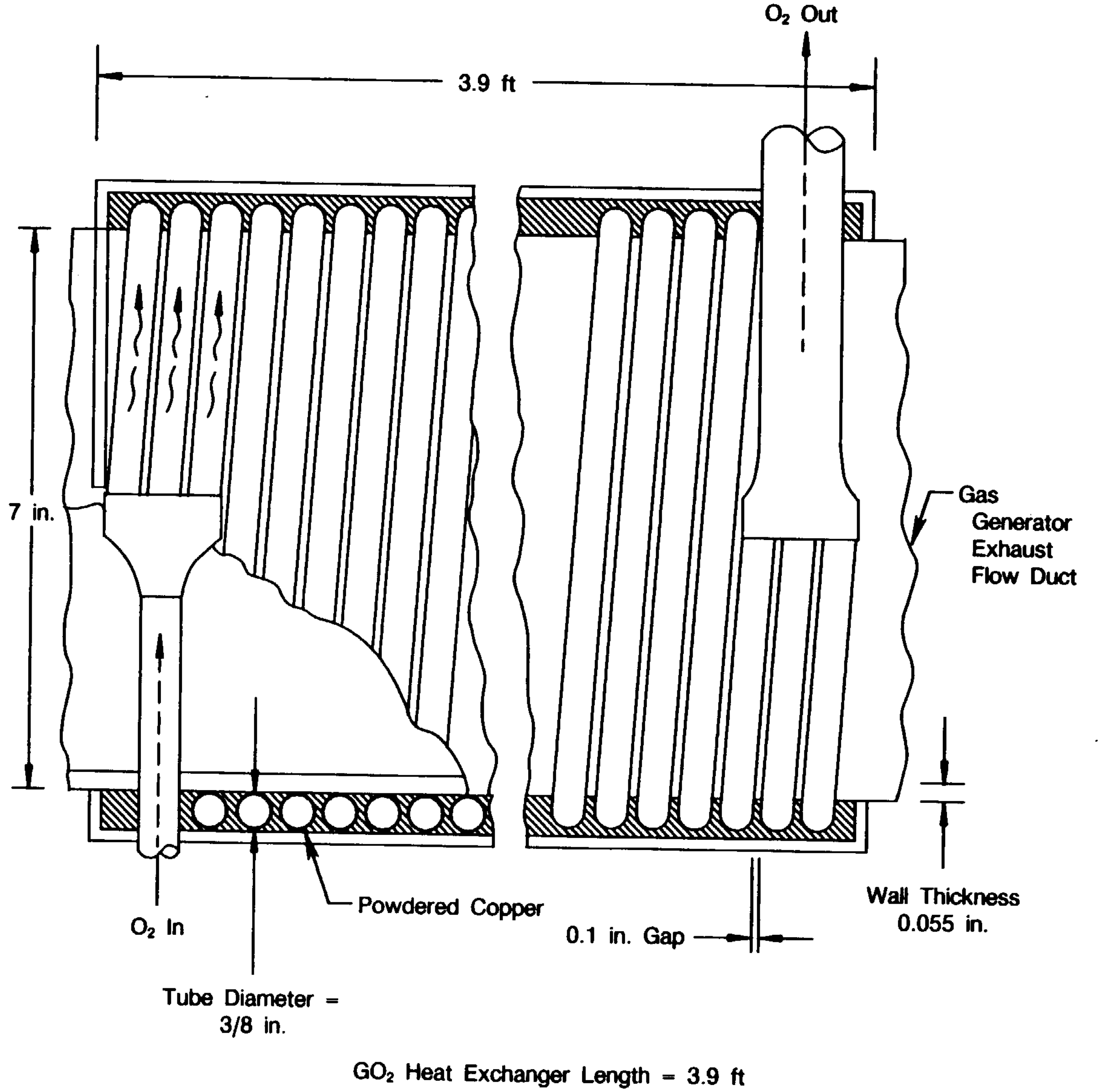

Figure 4.2.2.6-3. STBE Unique Split Expander GO_2 HEX Geometry and Performance Data

Table 4.2.2.6-1. Unique STBE Split Expander Performance — Design Power Level

Design Power Level	
Pressure — psia	895.3
Mixture Ratio	3.5
Area Ratio	13.5
Ideal I_{sp} — sec	342.2
Δ I_{sp} ERE — sec	−6.87
Δ I_{sp} KIN — sec	−0.65
Δ I_{sp} TDK — sec	−4.15
Δ I_{sp} BLM — sec	−1.24
Del. I_{sp} Vac — sec	329.3
Flowrate — lbm/sec	2597.3
Vacuum Thrust	855390.0

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Table 4.2.2.6-2. Unique STBE Split-Expander Performance — Normal Power Level

Normal Power Level	
Pressure — psia	764.5
Mixture Ratio	3.5
Area Ratio	13.5
Ideal I_{sp} — sec	341.9
Δ I_{sp} ERE — sec	−6.86
Δ I_{sp} KIN — sec	−0.74
Δ I_{sp} TDK — sec	−4.11
Δ I_{sp} BLM — sec	−1.28
Del. I_{sp} Vac — sec	328.9
Flowrate — lbm/sec	2220.8
Vacuum Thrust	730389.0

R19691/47

The engine Theoretical First Unit (TFU) production cost includes all the recurring operational production cost elements specified in the ALS engine WBS. It includes manufacturing and acceptance of the Integrated Engine System, System Engineering and Integration, Program Management, Facilities Maintenance and Tooling Maintenance. The TFU estimate is based on a lot size of 100 and a 90-percent learning curve.

The Operations Cost per launch per engine includes all costs associated with the operational flight program as described in the ALS engine WBS. It includes Program Management, System Engineering and Integration, Facilities Maintenance, Operation and Support, and Training. The Operations Cost is based on a flight rate of 10 missions per year and it is the estimated cost that will be achieved after 100 total missions have been flown.

4.3 UNIQUE TAP-OFF CYCLE ENGINE

4.3.1 Engine Cycle

The candidate STBE configuration studied during the Phase A contract is a tap-off cycle with liquid oxygen and liquid methane as propellants. This engine operates at a main chamber pressure of 2400 psia at the rated power level (RPL) of 750,000 pounds thrust. The engine has a fixed nozzle with an area ratio of 35:1 and delivers 305 seconds of sea level specific impulse at RPL. Table 4.3.1-1 presents selected engine characteristics at the rated power level.

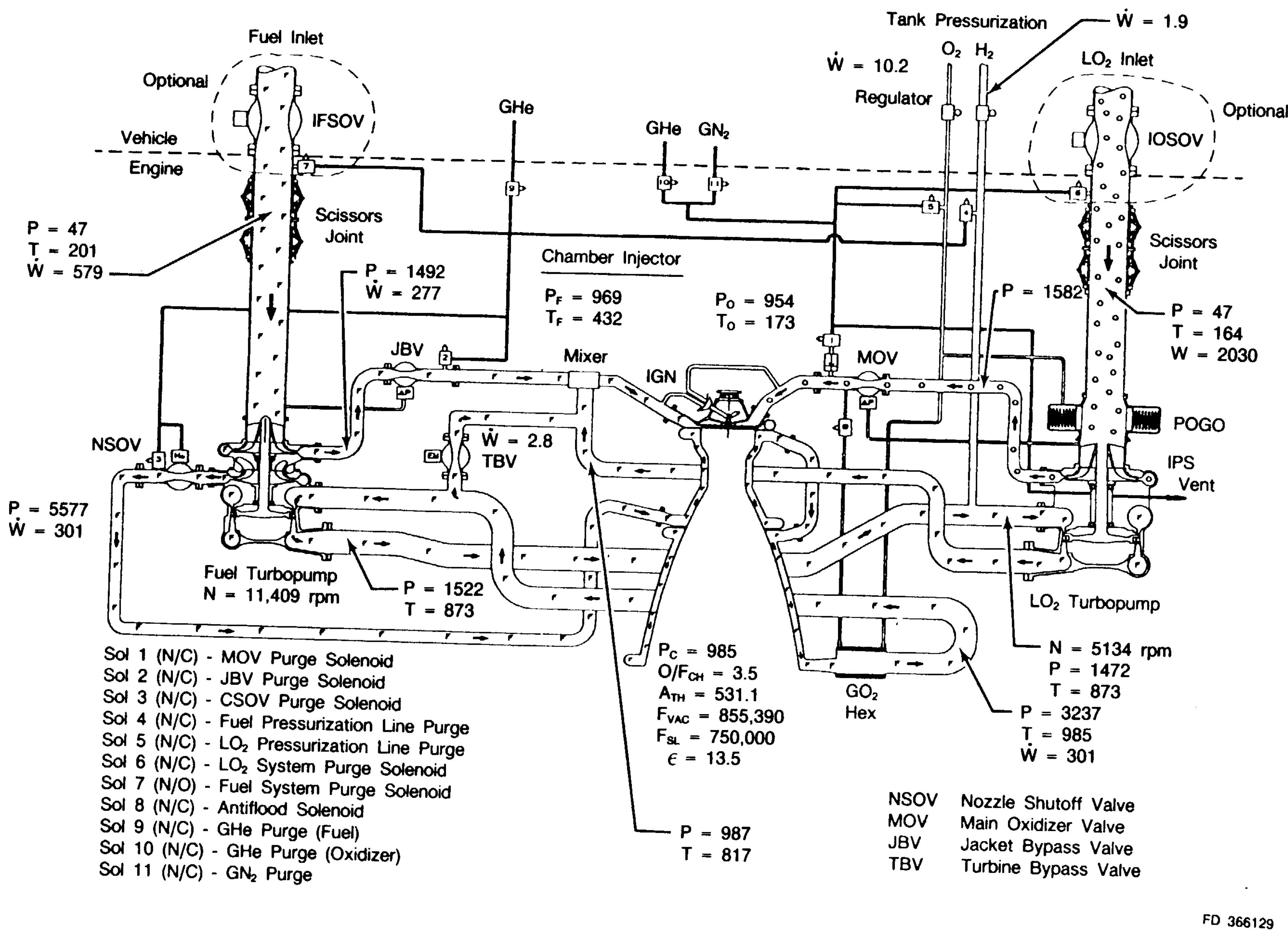

Figure 4.2.2.6-4. STBE Unique Split Expander Cycle Engine Operating Characteristics at Design Power Level

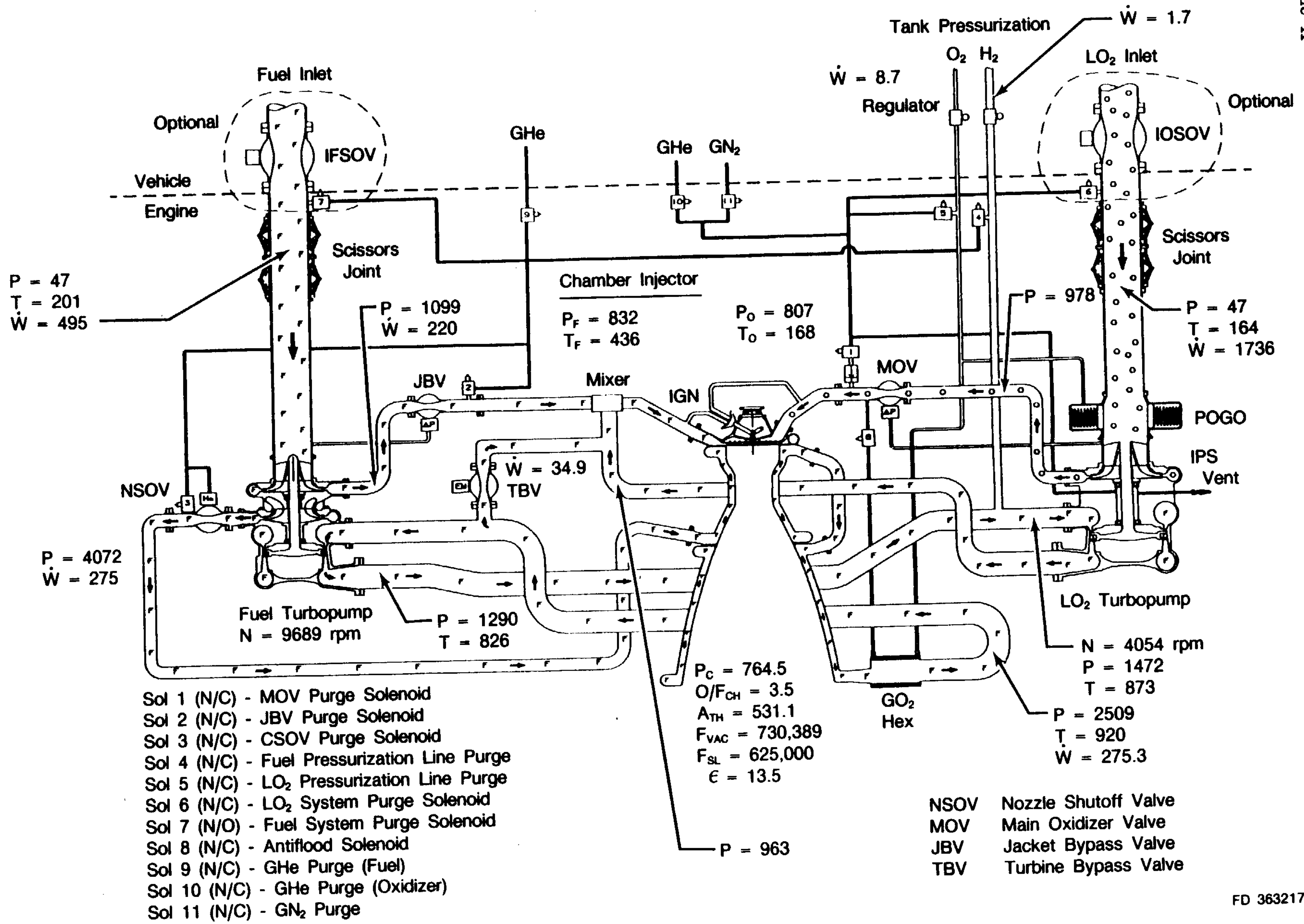

Figure 4.2.2.6-5. STBE Unique Split Expander Cycle Engine Operating Characteristics at Normal Power Level

Table 4.2.2.6-3. *Unique STBE Split Expander Engine Performance — Normal Power Level*

ENGINE PERFORMANCE PARAMETERS

CHAMBER PRESSURE	764.5
VAC ENGINE THRUST	730389.
S.L. ENGINE THRUST	625000.
TOTAL ENGINE FLOW RATE	2220.8
DEL. VAC. ISP	328.9
THROAT AREA	531.1
NOZZLE AREA RATIO	13.5
NOZZLE EXIT DIAMETER	95.5
NOZZLE LENGTH	94.91
ENGINE MIXTURE RATIO	3.50
CHAMBER COOLANT DP	1440.
CHAMBER COOLANT DT	687.
ETA C*	0.980
NOZZLE/CHAMBER Q	154994.

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	201.0	495.2	123.1	26.40
1ST STAGE EXIT	1098.6	206.7	495.2	132.4	26.52
JBV INLET	1075.6	206.8	219.9	132.4	26.51
JBV EXIT	963.0	207.4	219.9	132.4	26.44
PUMP EXIT	4072.1	233.7	275.3	167.4	26.42
NSOV INLET	4072.1	233.7	275.3	167.4	26.42
NSOV EXIT	3980.5	234.2	275.3	167.4	26.37
COOLANT INLET	3949.8	234.4	275.3	167.4	26.35
COOLANT EXIT	2509.4	920.2	275.3	730.3	3.98
TBV INLET	2480.7	920.0	34.9	730.3	3.94
TBV EXIT	963.0	905.4	34.9	730.3	1.59
CH4 TRB INLET	2480.7	920.0	240.4	730.3	3.94
CH4 TRB EXIT	1303.3	826.6	240.4	671.0	2.37
CH4 TRB DIFFUSER	1290.2	826.4	240.4	671.0	2.35
LOX TRB INLET	1267.8	826.0	240.4	671.0	2.31
LOX TRB EXIT	985.4	789.7	240.4	649.1	1.89
LOX TRB DIFFUSER	972.7	789.4	240.4	649.1	1.86
CH4 TANK OUT	963.0	789.3	1.7	649.1	1.85
CH4 TANK IN	47.0	770.8	1.7	649.1	0.09
GOX HEAT EXCH	963.0	789.7	238.7	649.1	1.89
MIXER	963.0	445.8	493.5	422.2	4.18
FSOV INLET	885.3	440.4	493.5	422.2	3.86
FSOV EXIT	840.5	437.0	493.5	422.2	3.67
CHAMBER INJ	832.0	436.4	493.5	422.2	3.63
CHAMBER	764.5				

* OXYGEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	164.0	1735.9	61.6	71.17
PUMP EXIT	978.0	167.8	1735.9	64.6	71.39
O2 TANK OUT	968.2	167.8	8.7	64.6	71.37
O2 TANK IN	47.0	400.0	8.7	204.3	0.36
OCV INLET	959.2	167.8	1727.3	64.6	71.36
OCV EXIT	814.0	168.3	1727.3	64.6	71.13
CHAMBER INJ	806.8	168.4	1727.3	64.6	71.12
CHAMBER	764.5				

* VALVE DATA *

VALVE	DELTA P	AREA	FLOW	% BYPASS
JBV	113.	6.02	219.88	44.40
TBV	1510.	1.11	34.91	12.68
NSOV	92.	8.37	275.33	
FSOV	45.	56.18	493.50	
OCV	145.	25.38	1727.26	

* INJECTOR DATA *

INJECTOR	DELTA P	AREA	FLOW	VELOCITY
FUEL	68.	47.15	493.50	415.21
LOX	42.	47.07	1727.26	74.30

Table 4.2.2.6-3. *Unique STBE Split Expander Engine Performance — Normal Power Level (Continued)*

TURBOMACHINERY PERFORMANCE DATA

CH4 TURBINE

EFFICIENCY (T/T)	0.867
EFFICIENCY (T/S)	0.843
SPEED (RPM)	9689.
MEAN DIA. (IN)	20.20
EFF AREA (IN2)	7.67
U/C (IDEAL)	0.462
MEAN TIP SPEED	855.
STAGES	1.
GAMMA	1.28
PRESS RATIO (T/T)	1.90
PRESS RATIO (T/S)	1.94
HORSEPOWER	20176.
EXIT MACH NUMBER	0.18

CH4 PUMP

	STAGE ONE	STAGE TWO
EFFICIENCY	0.787	0.594
HORSEPOWER	6538.	13638.
SPEED (RPM)	9689.	9689.
S SPEED	1351.	461.
HEAD (FT)	5708.	16231.
DIA. (IN)	14.29	21.60
TIP SPEED	605.	914.
VOL. FLOW	8380.	4678.
HEAD COEF	0.504	0.624
FLOW COEF	0.158	0.093

O2 TURBINE

EFFICIENCY (T/T)	0.875
EFFICIENCY (T/S)	0.795
SPEED (RPM)	4054.
MEAN DIA (IN)	23.35
EFF AREA (IN2)	16.40
U/C (IDEAL)	0.369
MEAN TIP SPEED	413.
STAGES	1.
GAMMA	1.28
PRESS RATIO (T/T)	1.29
PRESS RATIO (T/S)	1.32
HORSEPOWER	7460.
EXIT MACH NUMBER	0.20

O2 PUMP

EFFICIENCY	0.795
HORSEPOWER	7460.
SPEED (RPM)	4054.
S SPEED	1484.
HEAD (FT)	1880.
DIA. (IN)	19.94
TIP SPEED	353.
VOL. FLOW	10915.
HEAD COEF	0.486
FLOW COEF	0.170

Table 4.2.2.6-4. Unique STBE Split Expander Engine Performance — Design Power Level

ENGINE PERFORMANCE PARAMETERS
****

CHAMBER PRESSURE	895.3
VAC ENGINE THRUST	855390.
S.L. ENGINE THRUST	750000.
TOTAL ENGINE FLOW RATE	2597.3
DEL. VAC. ISP	329.3
THROAT AREA	531.1
NOZZLE AREA RATIO	13.5
NOZZLE EXIT DIAMETER	95.5
NOZZLE LENGTH	94.91
ENGINE MIXTURE RATIO	3.50
CHAMBER COOLANT DP	2232.
CHAMBER COOLANT DT	738.
ETA C*	0.980
NOZZLE/CHAMBER Q	178376.

ENGINE STATION CONDITIONS
****

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	201.0	579.1	123.1	26.40
1ST STAGE EXIT	1492.5	208.9	579.1	136.0	26.56
JBV INLET	1455.8	209.1	277.9	136.0	26.54
JBV EXIT	978.0	211.8	277.9	136.0	26.24
PUMP EXIT	5577.9	247.3	301.2	184.9	26.40
NSOV INLET	5577.9	247.3	301.2	184.9	26.40
NSOV EXIT	5506.4	247.7	301.2	184.9	26.36
COOLANT INLET	5469.6	247.9	301.2	184.9	26.34
COOLANT EXIT	3237.8	985.6	301.2	777.0	4.63
TBV INLET	3208.3	985.5	2.8	777.0	4.59
TBV EXIT	978.0	970.7	2.8	777.0	1.48
CH4 TRB INLET	3208.3	985.5	298.5	777.0	4.59
CH4 TRB EXIT	1522.8	873.4	298.5	702.6	2.60
CH4 TRB DIFFUSER	1504.5	873.3	298.5	702.6	2.57
LOX TRB INLET	1472.9	873.0	298.5	702.6	2.52
LOX TRB EXIT	1008.0	817.7	298.5	668.0	1.86
LOX TRB DIFFUSER	987.9	817.3	298.5	668.0	1.82
CH4 TANK OUT	978.0	817.2	1.9	668.0	1.80
CH4 TANK IN	47.0	800.3	1.9	668.0	0.09
GOX HEAT EXCH	978.0	817.7	296.5	668.0	1.86
MIXER	978.0	432.3	577.2	410.0	4.61
CHAMBER INJ	968.7	431.6	577.2	410.0	4.57
CHAMBER	895.3				

* OXYGEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	47.0	164.0	2030.3	61.6	70.98
PUMP EXIT	1582.5	170.6	2030.3	66.7	71.39
O2 TANK OUT	1566.6	170.7	10.2	66.7	71.37
O2 TANK IN	47.0	400.0	10.2	204.4	0.36
OCV INLET	1554.3	170.7	2020.1	66.7	71.35
OCV EXIT	963.7	172.9	2020.1	66.7	70.39
CHAMBER INJ	953.9	172.9	2020.1	66.7	70.38
CHAMBER	895.3				

* VALVE DATA *

VALVE	DELTA P	AREA	FLOW	% BYPASS
JBV	478.	3.69	277.85	47.98
TBV	2230.	0.07	2.77	0.92
NSOV	71.		301.25	
OCV	591.	14.72	2020.09	

* INJECTOR DATA *

INJECTOR	DELTA P	AREA	FLOW	VELOCITY
FUEL	73.	47.15	577.17	385.56
LOX	59.	47.07	2020.09	87.81

Table 4.2.2.6-4. *Unique STBE Split Expander Engine Performance — Design Power Level (Continued)*

PRATT & WHITNEY
STBE LOX/CH4 SPLIT EXPANDER ENGINE

TURBOMACHINERY PERFORMANCE DATA

CH4 TURBINE

EFFICIENCY (T/T)	0.883
EFFICIENCY (T/S)	0.857
SPEED (RPM)	11409.
MEAN DIA. (IN)	20.20
EFF AREA (IN2)	7.67
U/C (IDEAL)	0.489
MEAN TIP SPEED	1006.
STAGES	1.
GAMMA	1.26
PRESS RATIO (T/T)	2.11
PRESS RATIO (T/S)	2.16
HORSEPOWER	31443.
EXIT MACH NUMBER	0.20

CH4 PUMP

	STAGE ONE	STAGE TWO
EFFICIENCY	0.780	0.584
HORSEPOWER	10588.	20856.
SPEED (RPM)	11409.	11409.
S SPEED	1355.	447.
HEAD (FT)	7834.	22329.
DIA. (IN)	14.29	21.60
TIP SPEED	712.	1076.
VOL. FLOW	9785.	5121.
HEAD COEF	0.499	0.618
FLOW COEF	0.157	0.087

O2 TURBINE

EFFICIENCY (T/T)	0.877
EFFICIENCY (T/S)	0.796
SPEED (RPM)	5134.
MEAN DIA (IN)	23.35
EFF AREA (IN2)	16.40
U/C (IDEAL)	0.372
MEAN TIP SPEED	523.
STAGES	1.
GAMMA	1.26
PRESS RATIO (T/T)	1.46
PRESS RATIO (T/S)	1.52
HORSEPOWER	14615.
EXIT MACH NUMBER	0.26

O2 PUMP

EFFICIENCY	0.783
HORSEPOWER	14615.
SPEED (RPM)	5134.
S SPEED	1396.
HEAD (FT)	3102.
DIA. (IN)	19.94
TIP SPEED	447.
VOL. FLOW	12765.
HEAD COEF	0.500
FLOW COEF	0.157

Table 4.2.2.6-5. Unique STBE Gas Generator Costs

Total Development Cost (DDT&E), M$1010*
Production Cost (TFU), M$8.6
Operations Cost/Launch/Engine, M$0.128**
Constant FY87$

*Applies to developing a stand-alone booster engine configuration.
**Based on the 100th mission, 10 missions per year, and seven boosters per vehicle.

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Table 4.3.1-1. STBE Tap-Off Engine Characteristics — Rated Power Level

Performance	*Tap-Off*
Thrust - lb	750,000
Chamber Pressure - psia	2400
Mixture Ratio	3.0
Specific Impulse (Vac) - sec	342
Area Ratio	35

R19691/47

4.3.1.1 Flow Path Description

A simplified flow schematic for the STBE tap-off engine is presented in Figure 4.3.1-1 showing the major flow paths and components.

Liquid oxygen enters the engine at a net positive suction head (NPSH) level, supplied by the vehicle, sufficient for the high-speed high-pressure oxidizer pump. Liquid methane enters the engine at a NPSH level, again supplied by the vehicle, sufficient for the high-speed high-pressure methane pump, thus boost pumps are not required for this system.

At the rated power level, the methane pump operates at 16,295 rpm to provide the methane pressure level of 4368 psia required by the cycle. From the pump exit, the methane flows through the fuel shutoff valve where 85.7 percent of it flows to the inlet of the nozzle coolant passages. This methane regeneratively cools the tubular, stainless steel nozzle and milled channel, copper alloy main chamber. From here, the methane flows directly to the injector face. The remaining 12.5 percent of the methane flows through the fuel bypass valve and into the hot gas mixer.

The high-pressure oxidizer pump operates at 6,844 rpm to provide the oxygen pressure level of 3144 psia required by the cycle at the rated power level. From the pump exit, the oxygen flows through the main oxidizer control valve and is injected into the main chamber.

The tap-off provides 1.9 percent of the O/F biased chamber flow to the mixer inlet where cold methane mixes with the hot gases to provide 2293 psia, 1800 R gas to drive the high pressure propellant pumps. This mixed gas then flows through the hot gas valve to the inlet of the methane turbine. The hot gas is initially expanded through the methane turbine and is subsequently routed to a second turbine which powers the oxygen pump. The turbine exhaust gas is then expanded through an area ratio of 5:1 to atmospheric pressure providing additional thrust to the overall engine output.

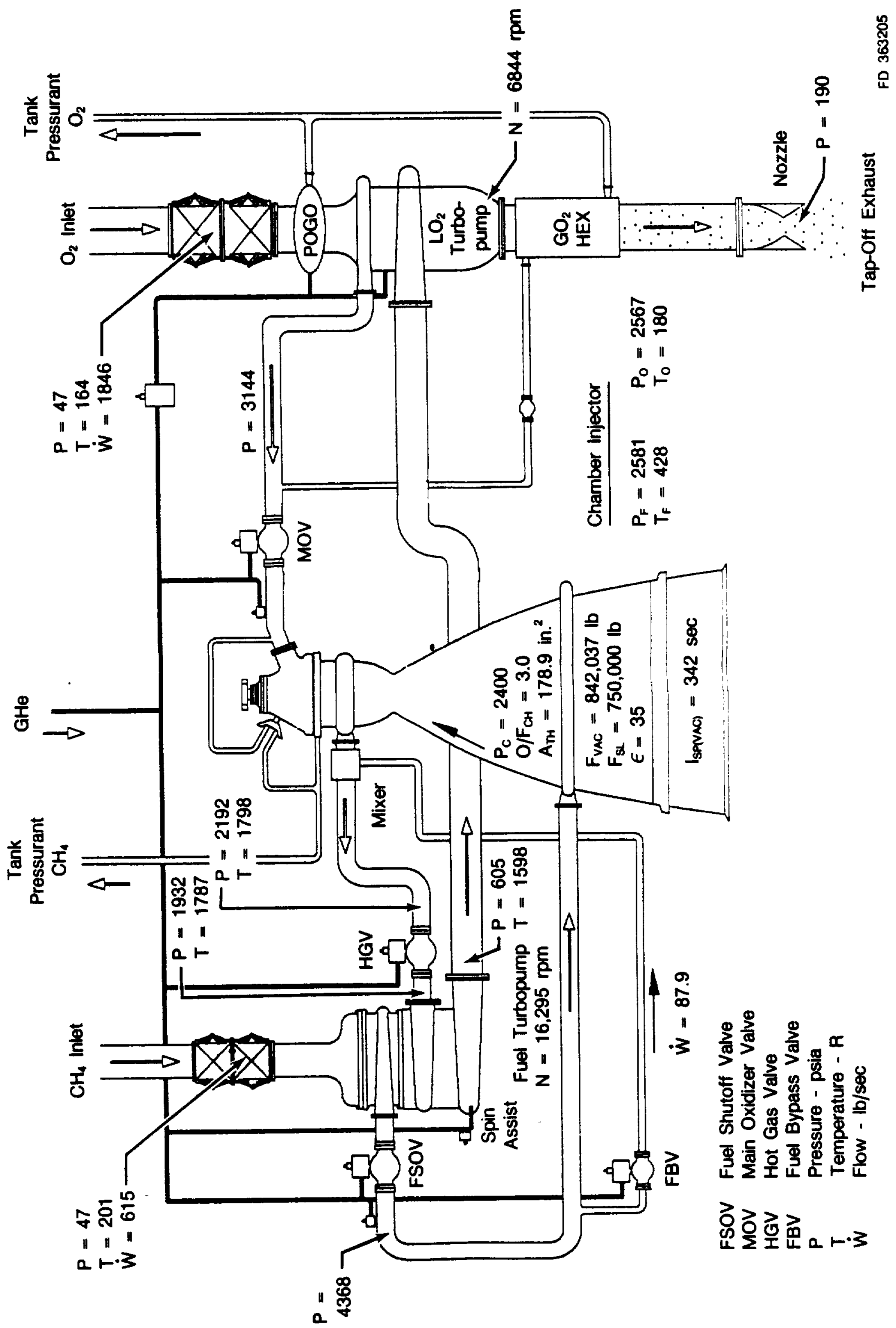

Figure 4.3.1-1. Simplified Flow Schematic for LO_2/CH_4 STBE Tap-Off Cycle Engine at Rated Power Level

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4.3.1.2 Engine Operation

The engine will be preconditioned using liquid flow from the tanks to soak the turbopumps until they are sufficiently cooled. The inlet valves will be opened, allowing liquid from the tanks to flow down to the turbopumps and letting any vapors percolate back up to the tank to be vented.

The engine start is a timed sequence process using an oxidizer lead for reliable soft propellant ignition. The oxidizer lead avoids hazardous buildup of unburned fuel in the combustor during the oxygen phase transition from gas to liquid. The transition occurs prior to fuel injection and the fuel is consumed immediately upon injection. Reliability of ignition is enhanced by the LO_2 lead because the transient mixture ratio during propellant filling includes the full excursion of ignitable mixture ratios from greater than 200 to less than one.

With the oxidizer lead sequence, the main chamber LO_2 injector is primed prior to opening the fuel shutoff valve to ensure liquid oxygen flow, eliminating turbine temperature spikes due to oxygen phase change. A helium spin assist is also used to initiate turbopump rotation before the fuel is introduced into the main chamber. During the start and shutdown, a small helium purge is used in the main chamber injector to eliminate the danger of hot gas flow reversals during transient operation. Main chamber ignition will be accomplished with dual electrical spark-excited, oxygen/methane torch igniters.

Main stage engine operation is open-loop controlled. The fuel bypass valve (FBV), the hot gas valve (HGV), and the main oxidizer valve (MOV), shown in Figure 4.3.1-1, are used to set the engine thrust and mixture ratio. Thrust and main chamber mixture ratio are set on the ground by trimming the HGV and MOV, respectively. The turbine inlet temperature is set using the FBV. All valves are operated by hydraulic actuators.

Engine acceleration is accomplished by a time-based scheduling of the valves to the commanded starting level (~ 20 percent power level). The acceleration to full thrust is also accomplished with open-loop valve schedules. Engine shutdown is accomplished using a time-based scheduling of the propellant valves. The HGV is closed first to power down the turbopumps, then the MOV closes, followed by shutting off the methane system.

In addition to a normal operational mode, the engine system is capable of shutdown resulting from detected problems or LO_2 starvation at the end of the burn duration.

4.3.1.3 Combustor

4.3.1.3.1 Thrust Chamber and Nozzle Cooling

A preliminary cooling system for the thrust chamber and nozzle for a tap-off cycle has been configured and its thermal characteristics predicted. The configuration has an injector diameter of 23.9 inches, a throat diameter of 15.1 inches, with a corresponding contraction ratio of 2.5:1. The thrust chamber/nozzle assembly is 80-inches long and extends to a nozzle expansion ratio of 35:1. The thrust chamber has an integral acoustic liner, extends to a nozzle area ratio of 4.0:1 and features a mechanical passage thermal-skin NASA-Z/nickel closeout assembly surrounded by a structural jacket. The regeneratively cooled nozzle mates with the thrust chamber at an area ratio of 4.0:1, is 43-inches long and is fabricated from 810 single taper, constant wall thickness Haynes 230 tubes. The required tap-off flow is bled from the acoustic cavity which surrounds the forward portion of the thrust chamber. The injector is heavily O/F biased to provide a wall O/F of 2.0 which reduces the thermal severity to the forward portion of the chamber. The O/F ratio of the core flow from the injector is adjusted to provide an overall O/F of 3.5.

A counterflow cooling system that flows all of the methane fuel is used for both the nozzle and thrust chamber. The coolant enters the tube assembly, transitions into the machined passages of the thrust chamber, flows through the chamber/acoustic liner assembly and discharges into the injector. The critical coolant passage dimensions are sized to meet the heat transfer cycle requirements at the 750K lbf sea level thrust at 2400 psia chamber pressure design point and reflect the following design guidelines:

- Thrust chamber liner wall thickness > 0.030 inch.
- Machined passage aspect ratio < 5.0.
- Machined passage land width > 0.050 inch.
- Cooling enhancement from passage curvature.
- Coolant Mach number < 0.5.
- Tube wall thickness > 0.013 inch.
- Tubular stress < 0.2 percent yield stress.
- Ultimate tube temperature margin > 375 R.

The methane enters the coolant system at 239 R and 5055 psia and exits into the injector at 450 R and 2607 psia. Table 4.3.1-2 summarizes the predicted thermal operating characteristic of the cooling system. Recent analysis has increased the number of coolant tubes from 540 to 810 to improve structural margin. The higher coolant pressure loss associated with this increase in tube number is reflected in the table but may not be shown in the performance cycle.

Table 4.3.1-2. *STBE Tap-Off Gas Generator Cycle Performance*

Fuel	CH_4
Overall O/F Ratio	3.50
Sea Level Thrust (lbf)	750,000
Chamber Pressure (psia)	2,400
Throat Area (in.^{2})	178.9
Injector Flow Rate (lbm/sec)	2,462
Throat Flow Rate (lbm/sec)	2,329
Tap-Off Flow Rate (lbm/sec)	132
Coolant Flow Rate (lbm/sec)	528
Exit Area Ratio	35
Coolant Flow Rate (lbm/sec)	528
Inlet Pressure (psia)	5,055
Pressure Drop (psid)	2,448
Exit Pressure (psia)	2,607
Inlet Temperature (deg R)	239
Temperature Rise (deg R)	211
Exit Temperature (deg R)	450
Total Heat Pickup (Btu/sec)	97,342

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4.3.1.4 Engine Costs

This section summarizes cost estimates for the 750K SL thrust, 2400 psia chamber pressure, Tap-Off STBE Cycle. Table 4.3.1-3 summarizes significant costs for the engine.

Table 4.3.1-3. *Tap-Off Cycle Engine Costs*

Total Development Cost (DDT&E), M$1400*
Production Cost (TFU), M$11.1
Operations Cost/Launch/Engine, M$0.155**
Constant FY87$

*Applies to developing a stand-alone booster engine configuration.
**Based on the 100th mission, 10 missions per year, and seven boosters per vehicle.

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The DDT&E Cost includes all of the functions required to design, develop, test and evaluate the engine system. All of the DDT&E functions shown in the ALS engine WBS (see Volume III) have been included. Development Cost is based on a 90-month phase C/D program with 960 engine firings for the tap-off STBE. Sufficient accountable firings have been included in the program to demonstrate 0.99 engine reliability with one failure.

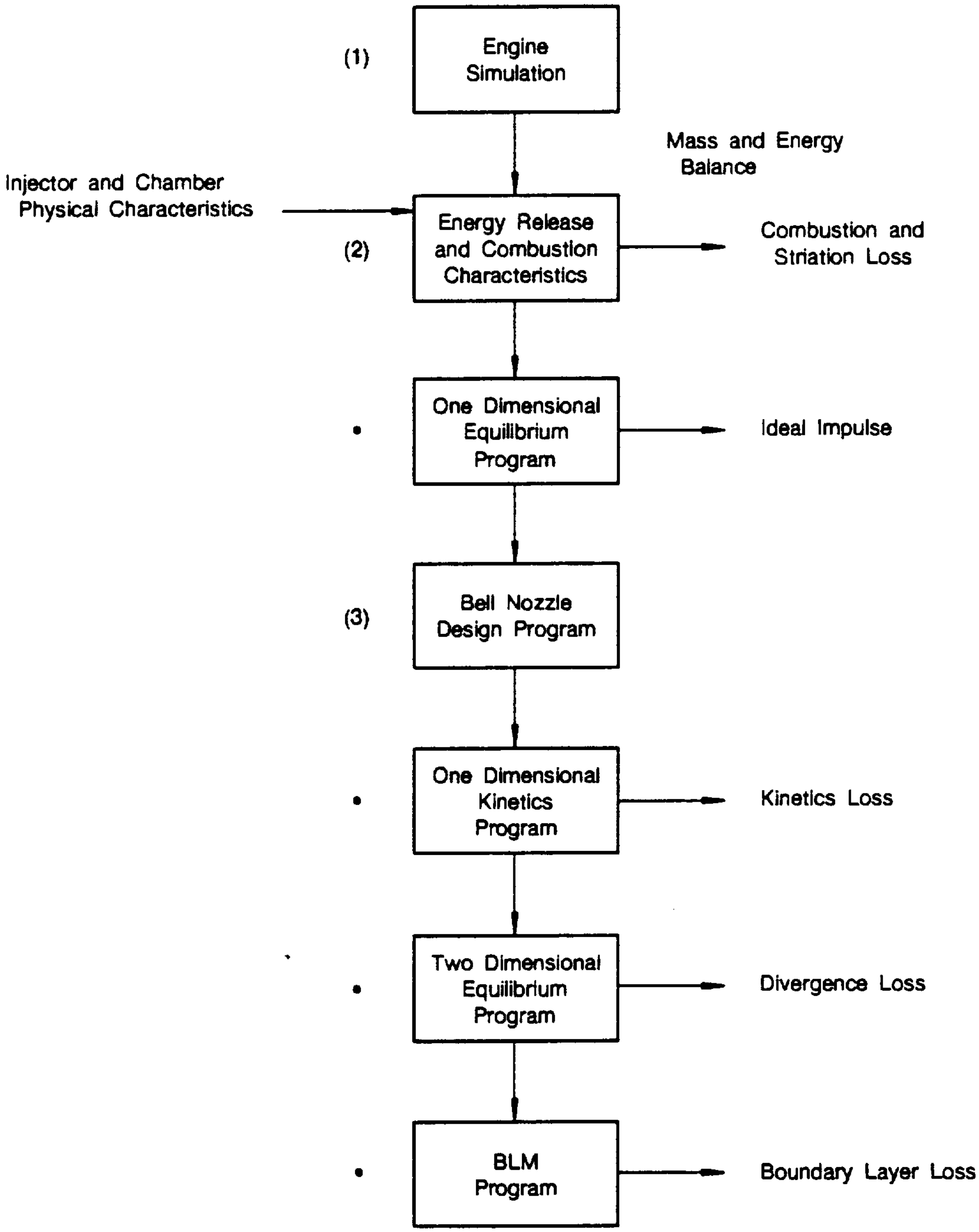
The engine Theoretical First Unit (TFU) production cost includes all the recurring operational production cost elements specified in the ALS engine WBS. It includes manufacturing and acceptance of the Integrated Engine System, System Engineering and Integration, Program Management, Facilities Maintenance and Tooling Maintenance. The TFU estimate is based on a lot size of 100 and a 90-percent learning curve.

The Operations Cost per launch per engine includes all costs associated with the operational flight program as described in the ALS engine WBS. It includes Program Management, System Engineering and Integration, Facilities Maintenance, Operation and Support, and Training. The Operations Cost is based on a flight rate of 10 missions per year and it is the estimated cost that will be achieved after 100 total missions have been flown.

4.3.2 Engine Performance

The STBE tap-off system performance was determined during the preliminary design using the accepted JANNAF methodology. Rigorous procedures have been established for use in calculating chamber/nozzle thrust and specific impulse. The steady-state design point computer simulation provided an initial match of components and definitions of mixture ratio, mass flow, temperature and pressure levels for the detailed performance calculations using the JANNAF methodology. Figure 4.3.2-1 shows a flow schematic of the JANNAF performance prediction procedure followed during this Task. Performance was estimated for both the main chamber flow and the Tap-Off, which is dumped overboard during engine operation. Table 4.3.2-1 lists the detailed performance estimates at the rated power level (RPL) thrust of 750,000 pounds. Overall engine performance was calculated by mass weighing the main chamber flow performance with the Tap-Off flow performance

During this study, detailed aerothermal analyses were made to predict component performance levels and these were incorporated into a steady-state computer model of the complete engine. A simplified flow schematic is presented in Figure 4.3.1-1 with key operating parameters noted. Table 4.3.2-2 defines performance of the individual components and their operating environment for the STBE Tap-Off at RPL.



• Designates JANNAF Computer Programs
(1) Engine Steady-State Computer Program
(2) Predicted Using Techniques and Programs Developed During Previous Rocket Engine Programs
(3) Program Developed at P&W for Bell Nozzle Design Uses Method of Characteristics Calculations

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Figure 4.3.2-1. Performance Prediction Procedure

Table 4.3.2-1. *STBE Tap-Off Gas Generator Cycle Performance*

Fuel	CH_4
Overall O/F Ratio	3.50
Sea Level Thrust-lbf	750,000
Chamber Pressure-psia	2,400
Throat Area-in.^{2}	178.9
Injector Flow Rate-lbm/sec	2,462
Throat Flow Rate-lbm/sec	2,329
Tap-Off Flow Rate-lbm/sec	132
Coolant Flow Rate-lbm/sec	528
Exit Area Ratio	35
Coolant Flow Rate-lbm/sec	528
Inlet Pressure-psia	5,055
Pressure Drop-psid	2,448
Exit Pressure-psia	2,607
Inlet Temperature-R	239
Temperature Rise-R	211
Exit Temperature-R	450
Total Heat Pickup-Btu/sec	97,342

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Table 4.3.2-2. STBE Tap-off Engine Performance — Rated Power Level

* PRATT & WHITNEY *
* TAPOFF CYCLE DESIGN DECK *

ENGINE PERFORMANCE	
VACUUM THRUST	842037.
SEA LEVEL THRUST	750001.
VACUUM IMPULSE	342.04
SEA LEVEL IMPULSE	304.66
TOTAL ENGINE INLET FLOW RATE	2461.8
OVERALL ENGINE MIXTURE RATIO	3.00

ENGINE HEAT TRANSFER	
NOZZLE/CHAMBER COOLANT DP	1546.
NOZZLE/CHAMBER COOLANT DT	199.
NOZZLE/CHAMBER Q	91379.

CHAMBER PERFORMANCE	
PRESSURE	2400.0
TEMPERATURE	6608.6
THRUST	820249.
IMPULSE	352.14
THROAT FLOW RATE	2329.3
THROAT AREA	178.93
NOZZLE AREA RATIO	35.
MIXTURE RATIO	3.50
NOZZLE EFFICIENCY	0.980
CSTAR EFFICIENCY	0.980

TAPOFF PERFORMANCE	
MIXER PRESSURE	2293.2
MIXER TEMPERATURE	1800.0
THRUST	21788.
IMPULSE	164.50
FLOW RATE	132.5
NOZZLE AREA	69.69
NOZZLE AREA RATIO	5.
MIXTURE RATIO	0.354
NOZZLE EFFICIENCY	0.980

ENGINE STATION CONDITIONS

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	201.0	615.4	123.1	26.40
MAIN PUMP EXIT	4367.6	228.4	615.4	164.6	26.73
MFV INLET	4280.3	228.9	615.4	164.6	26.68
MFV EXIT	4237.4	229.2	615.4	164.6	26.66
NZ/CH COOL INLET	4152.7	229.7	527.5	164.6	26.61
NZ/CH COOL EXIT	2606.7	428.6	527.5	337.8	14.48
CH INJ INLET	2580.6	428.2	527.5	337.8	14.43
FBV INLET	4152.7	229.7	87.9	164.6	26.61
FBV EXIT	2400.0	239.1	87.9	164.6	25.54
MIXER INLET	2352.0	239.4	87.9	164.6	25.51

* OXIDIZER SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
MAIN PUMP INLET	47.0	164.0	1846.3	61.6	71.17
MAIN PUMP EXIT	3144.3	178.0	1846.3	72.2	71.69
MOV INLET	3081.4	178.3	1846.3	72.2	71.60
MOV EXIT	2619.2	180.2	1846.3	72.2	70.90
CH INJ INLET	2566.8	180.4	1846.3	72.2	70.82

* HOT GAS SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW
CHAMBER TAPOFF	2400.0	4918.5	44.5
MIXER INLET	2352.0	4918.1	44.5
MIXER EXIT	2235.9	1798.1	132.5
HGV INLET	2191.2	1796.6	132.5
HGV EXIT	1972.0	1788.8	132.5
FUEL TURB INLET	1932.6	1787.3	132.5
FUEL TURB EXIT	605.3	1598.9	132.5
LOX TURB INLET	575.0	1595.6	132.5
LOX TURB EXIT	200.0	1447.3	132.5
NOZZLE INLET PRES	190.0		

* PRATT & WHITNEY *
* TAPOFF CYCLE DESIGN DECK *

TURBOMACHINERY PERFORMANCE DATA

* FUEL TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.836	0.820
HORSEPOWER	18068.	18068.
SPEED (RPM)	16295.	16295.
S SPEED	35.1	45.0
S DIAMETER	1.91	1.50
MEAN DIAMETER (IN)	14.43	14.43
VEL.RATIO (ACTUAL)	0.47	0.47
MAX TIP SPEED	1080.	1105.
BLADE HEIGHT (IN)	0.75	1.10
AN SQUARED	90.0	131.9
EFFECTIVE AREA	7.72	13.03
PRES.RATIO (T/T)	1.74	1.83
GAS CONSTANT (FT)	93.98	
GAMMA	1.1624	

* FUEL PUMP *

EFFICIENCY	0.723
HORSEPOWER	36136.
SPEED (RPM)	16295.
NPSH (FT)	177.7
SS SPEED	42175.
S SPEED	883.
HEAD (FT)	23335.
DIAMETER (IN)	16.37
TIP SPEED (FT/SEC)	1165.
VOL FLOW	10464.
HEAD COEF	0.5535
FLOW COEF	0.0690

* LOX TURBINE *

	STAGE ONE	STAGE TWO
EFFICIENCY (T/T)	0.806	0.765
HORSEPOWER	13852.	13852.
SPEED (RPM)	6844.	6844.
S SPEED	28.3	34.7
S DIAMETER	1.71	1.38
MEAN DIAMETER (IN)	22.63	22.63
VEL.RATIO (ACTUAL)	0.35	0.35
MAX TIP SPEED	720.	737.
BLADE HEIGHT (IN)	1.46	2.03
AN SQUARED	48.7	67.6
EFFECTIVE AREA	24.44	38.91
PRES.RATIO (T/T)	1.64	1.75
GAS CONSTANT (FT)	93.15	
GAMMA	1.1710	

* LOX PUMP *

EFFICIENCY	0.755
HORSEPOWER	27705.
SPEED (RPM)	6844.
NPSH (FT)	62.4
SS SPEED	35971.
S SPEED	1053.
HEAD (FT)	6231.
DIAMETER (IN)	20.50
TIP SPEED (FT/SEC)	613.
VOL FLOW	11645.
HEAD COEF	0.5339
FLOW COEF	0.0839

* VALVE DATA *

STATION	DELP	AREA	FLOW	%DELP/P
FUEL SHUT OFF VLV	42.8	27.25	615.4	1.00
MAIN OXID VALVE	462.2	15.18	1846.3	15.00
FUEL BYPASS VALVE	1752.7	0.609	87.9	42.21
HOT GAS VALVE	219.1	10.47	132.5	10.00

* INJECTOR DATA *

STATION	DELP	AREA	FLOW	%DELP/P
FUEL CH INJ	180.6	20.00	527.5	7.00
LOX CH INJ	166.8	25.41	1846.3	6.50

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2. Rosner, D. E., On Liquid Droplet Combustion at High Pressures; AIAA Journal, Vol. 5, No. 1, pp. 163-166, January, 1967.

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SECTION 5.0
STBE PROGRAMMATIC ANALYSES AND PLANS

Introduction

This section describes the work conducted under Task V (SOW Task 5.5). It describes the development plan for the Derivative STBE Gas Generator Engine following the ground rules established by NASA in late 1988 and as summarized in a NASA DDT&E ground rule document dated 20 December 1988. The basic requirement is for a 90-month DDT&E program through Final Flight Certification for an STME engine and an STBE engine derived from the STME.

The objective of the STME DDT&E program is to develop a 580,000-pound vacuum thrust LO_2/LH_2 rocket engine to be used on the core vehicle. The derivative STBE engine is to be a LO_2/CH_4 rocket engine which uses as much hardware common to the STME engine as possible. The resulting derivative STBE has a vacuum thrust of 706.5K pounds and sea level thrust of 500K pounds. Seven derivative STBE engines are to be used on the booster and three engines on the core vehicle (for the purposes of the development plan).

Milestone Dates

The milestone dates as specified by NASA and shown in Table 5.0-1 were used to develop the DDT&E plan.

Table 5.0-1. *STME DDT&E Milestone Dates*

Date	*Milestone*
Jan. 1989	Start Advanced Development Program for gas generator, thrust chamber, turbopumps and engine controls.
June 1989	Start STME Phase B
Oct. 1991	Start Full-Scale Development
Oct. 1993	Component and Subsystem Development Test Facility (CSDTF) available
June 1994	First LO_2/LH_2 engine stand available — 2 positions
Sept. 1994	First LO_2/CH_4 engine stand available — 2 positions
Oct. 1994	Two additional test stands available — 2 positions
Aug. 1995	Critical Design Review
Sept. 1996	MPTA stand available (cluster test)
July 1997	Complete Preliminary Flight Certification, deliver first flight engine set with three spare engines
Jan. 1998	Deliver second flight engine set with three spare engines
Apr. 1998	First flight
Oct. 1998	Second flight
Mar. 1999	Complete Final Flight Certification Tests

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DDT&E Ground Rules

A series of ground rules as specified by NASA, and additional P&W ground rules were used to establish the development plan. These ground rules are shown in Table 5.0-2.

Included in the following sections are: the logic network; the schedules; the test facility requirements; and the Environmental Analysis (DR-10). The Work Breakdown Structure (WBS) and program cost estimates are contained in Volume III of this report.

5.1 LOGIC NETWORK

The logic network shown in Figure 5.1-1 is distributed in time phases, starting with Phase A, Technology and Concept Development, and extending through Production. The items addressed to the appropriate depth for each phase are:

- Chamber/injector demonstration
- Engine design, testing, and production
- Facilities, tooling, and special test equipment
- Launch and flight support.

The Phase A items are described throughout this report and each item is addressed in some detail. Phase A should lead into a Phase A' where more detail will be put into the engine design and analysis. The greater level of detail in Phase A' will allow the various plans to be formulated, along with the very critical safety analyses.

One of the items in Phase B is a demonstration of the combustion efficiency, combustion stability, and heat transfer in tests of a full-scale chamber and injector.

An engine Preliminary Design Review (PDR) will be conducted in Phase B. This design review will be made as a result of the design and analysis that supports engineering layout drawings of the selected concept. At this point, the definition of the engine is sufficiently complete to allow all of the items that were previously labeled preliminary to be finalized. This will also allow the creation of the Design Verification and Substantiation (DVS) requirements for the engine components. The chamber and injector DVS requirements can be used to formulate the test plan for the demonstration chamber and injector.

Completion of the engine layout drawings for PDR allows the planning for the support items to be done. This includes the ground support equipment, tooling, operation, and maintenance planning.

At this point, enough definition of the program has been generated to allow the preparation of a comprehensive Phase C/D proposal.

As the program progresses into Phase C/D, the layout drawings can be turned into detail fabrication drawings. The drawings will be used to fabricate the components and to conduct a comprehensive Critical Design Review (CDR). During fabrication and at fabrication completion, the various component parts and assemblies will be subjected to the DVS tests per the DVS plans that were created during Phase B. The same applies to parts necessary for the engine assembly level, such as flow ducting.

Table 5.0-2. *STME/Derivative STBE Development Ground Rules*

NASA Groundrules

1. 90-month program through FFC
2. Flight Qualified Engine Life — 15 missions
3. STME engine is to be used for core. Derivative STBE is to be used for the booster stage.
4. 0.99 minimum demonstrated reliability at 90 percent confidence prior to first flight for both engines.
5. Component and engine test conducted by P&W at government owned and operated test facilities at Stennis Space Center. The government will maintain the test facilities down to the interface connections with the test article.
6. The government is to supply the propellants and pressurants at no charge to the contractor.
7. 960 total engine firings through flight testing and final flight certification — applies to the STME. (P&W has established derivative STBE requirement at 488 total engine firings).
8. Two flight tests of booster and core vehicle from ESMC
9. Booster engines are recovered and refurbished following flight test. Core engines are expended.
10. Flight and MPTA engine spares — one spare engine for every three delivered engines.

Additional P&W Ground Rules

1. 488 Derivative STBE engine firings selected for development requirement and to meet reliability requirement of 0.99 at 90 percent confidence on the derivative STBE.
2. STME design, fabrication and testing lead the derivative STBE.
3. Design verification tests on the same or similar STME/Derivative STBE component will be conducted with the higher load set.
4. Conduct verification test with CH_4 on common parts.
5. Hardware design life: 120 firings

   Maximum firing on development hardware: 60 firings.

   Maximum tests between overhauls: 30 firings.
6. Rig mount time (GG and pumps)

with minimal instrumentation:	1 week *
with extensive instrumentation:	2 weeks *
Rig dismount time:	1 week

   * Add one week for main combustion chamber rig.
7. Engine mount time:

with minimal instrumentation:	1 week
with extensive instrumentation:	2 weeks
Engine dismount time:	1 week

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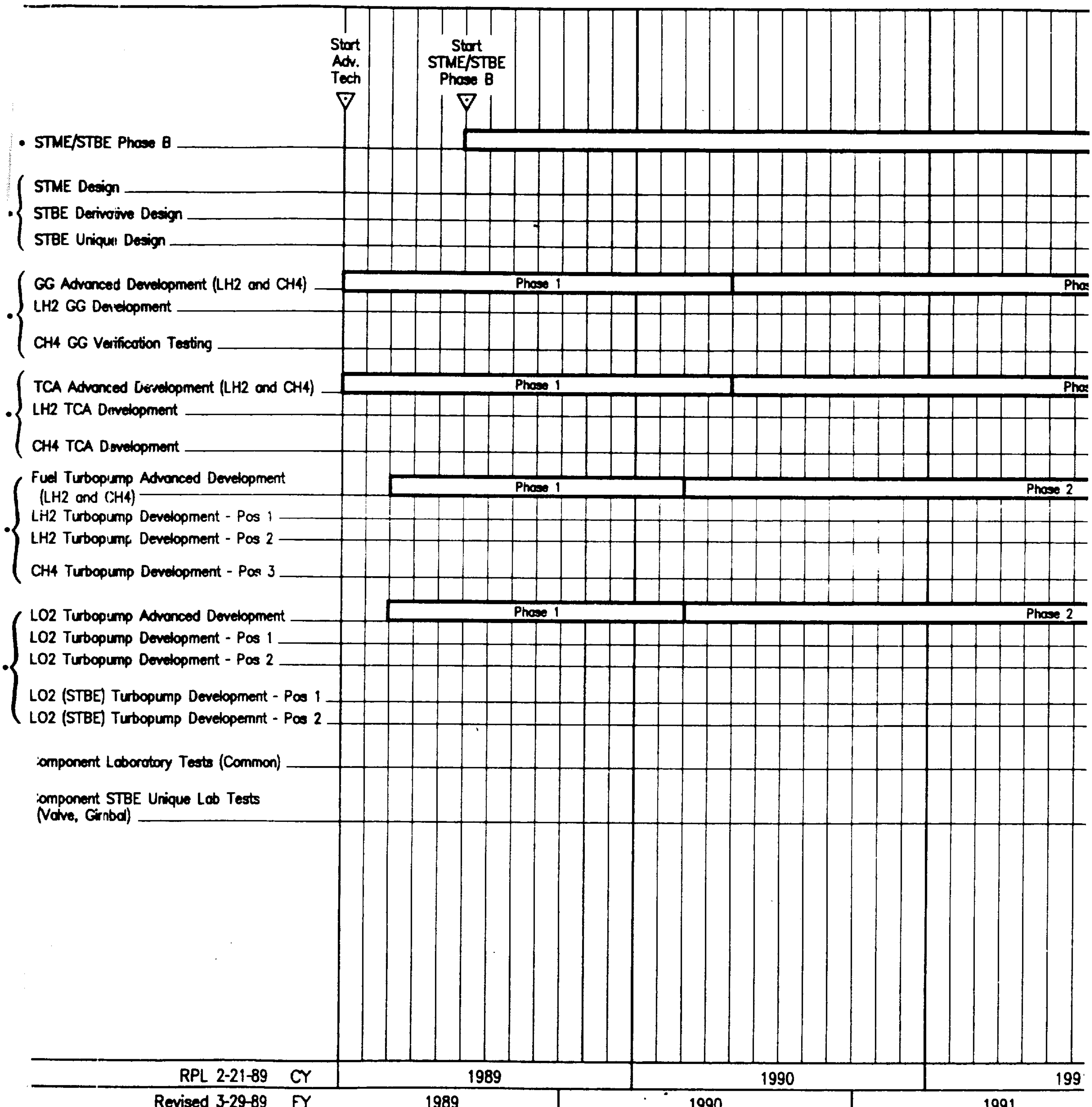

ure 5.2-1. STME/Derivative STBE Development Schedule (Sheet 3 of 4)

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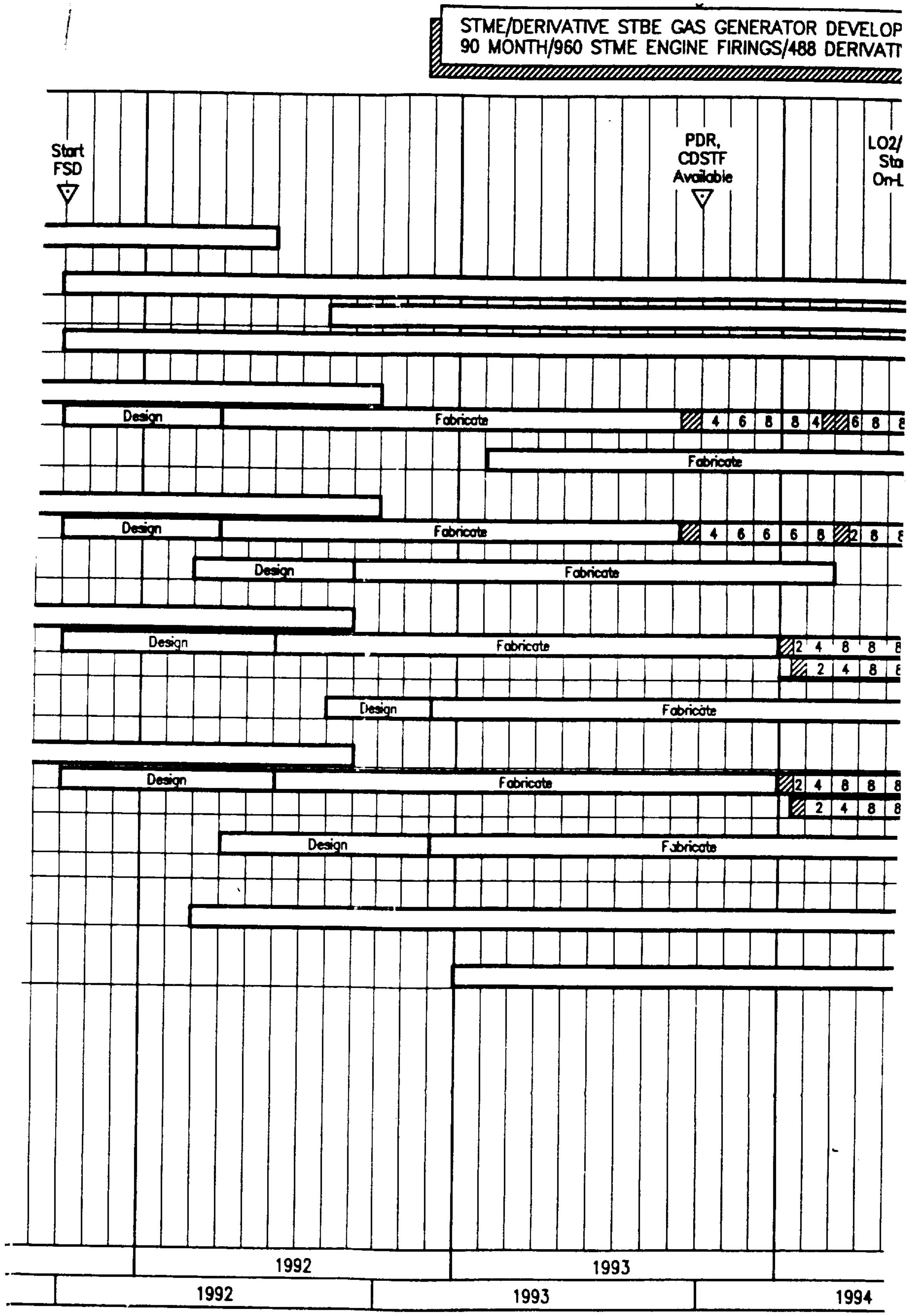

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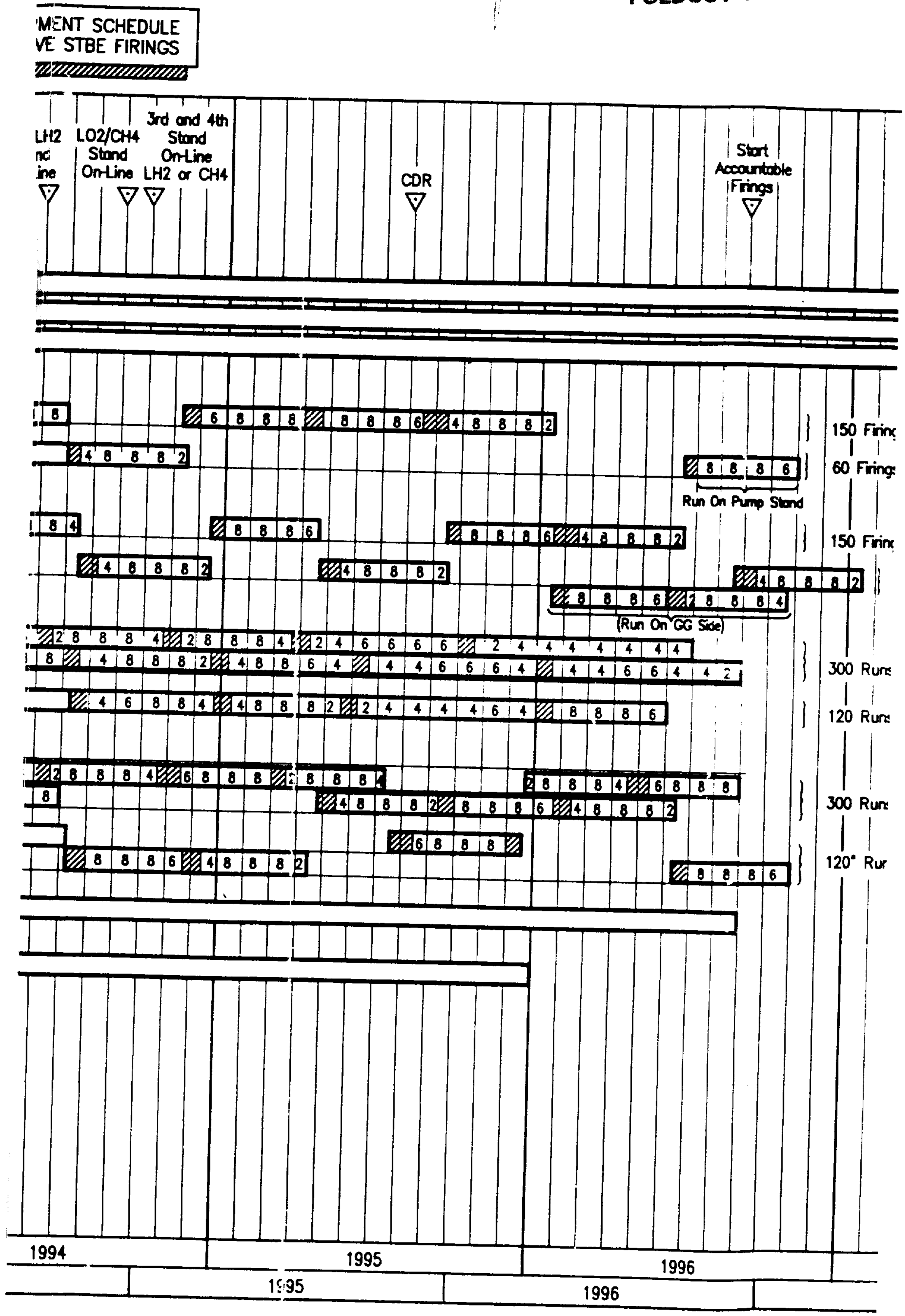

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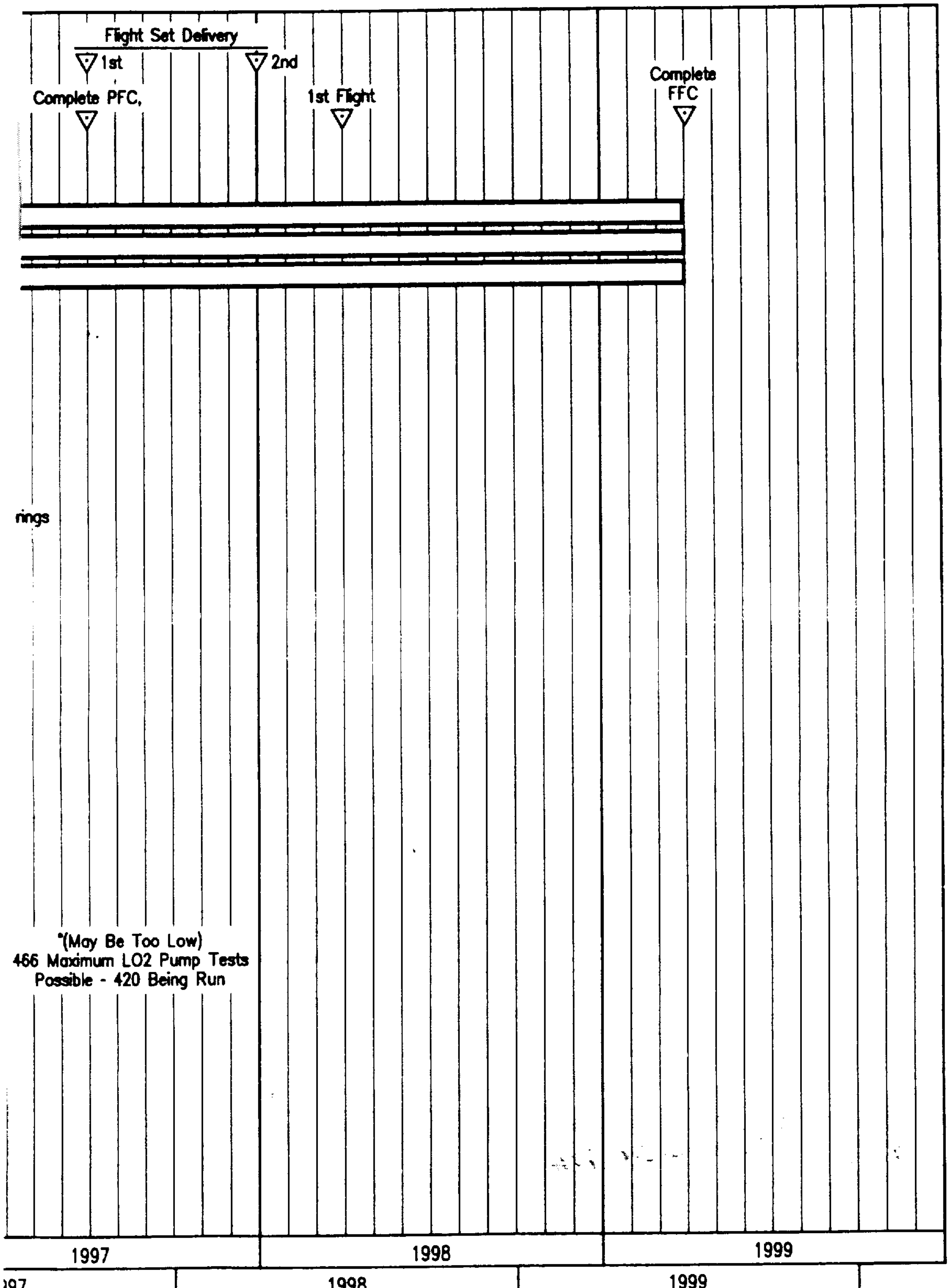

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- STME/STBE Phase
- Engine Design (Ref)

- Engine Test Stand
  Engine Test Stand

- Engine Test Stand
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- Engine Tests Stan
  Engine Tests Stan

- Engine Test Stand
  Engine Test Stand

- MPTA, Flight Tests

Number of STME Engi
Cumulative STME Engi

Number of STME Acc
Cumulative STME Acc

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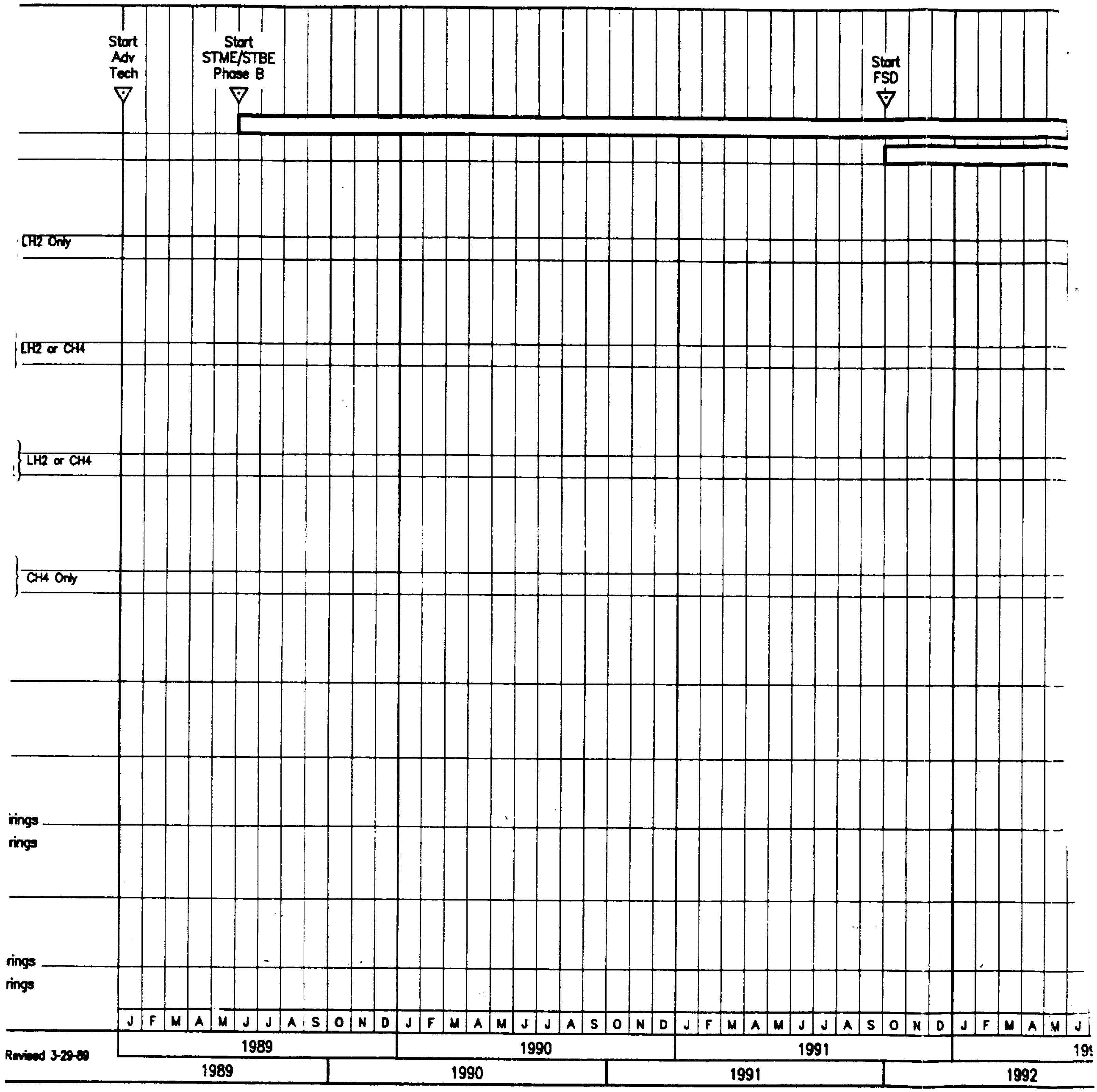

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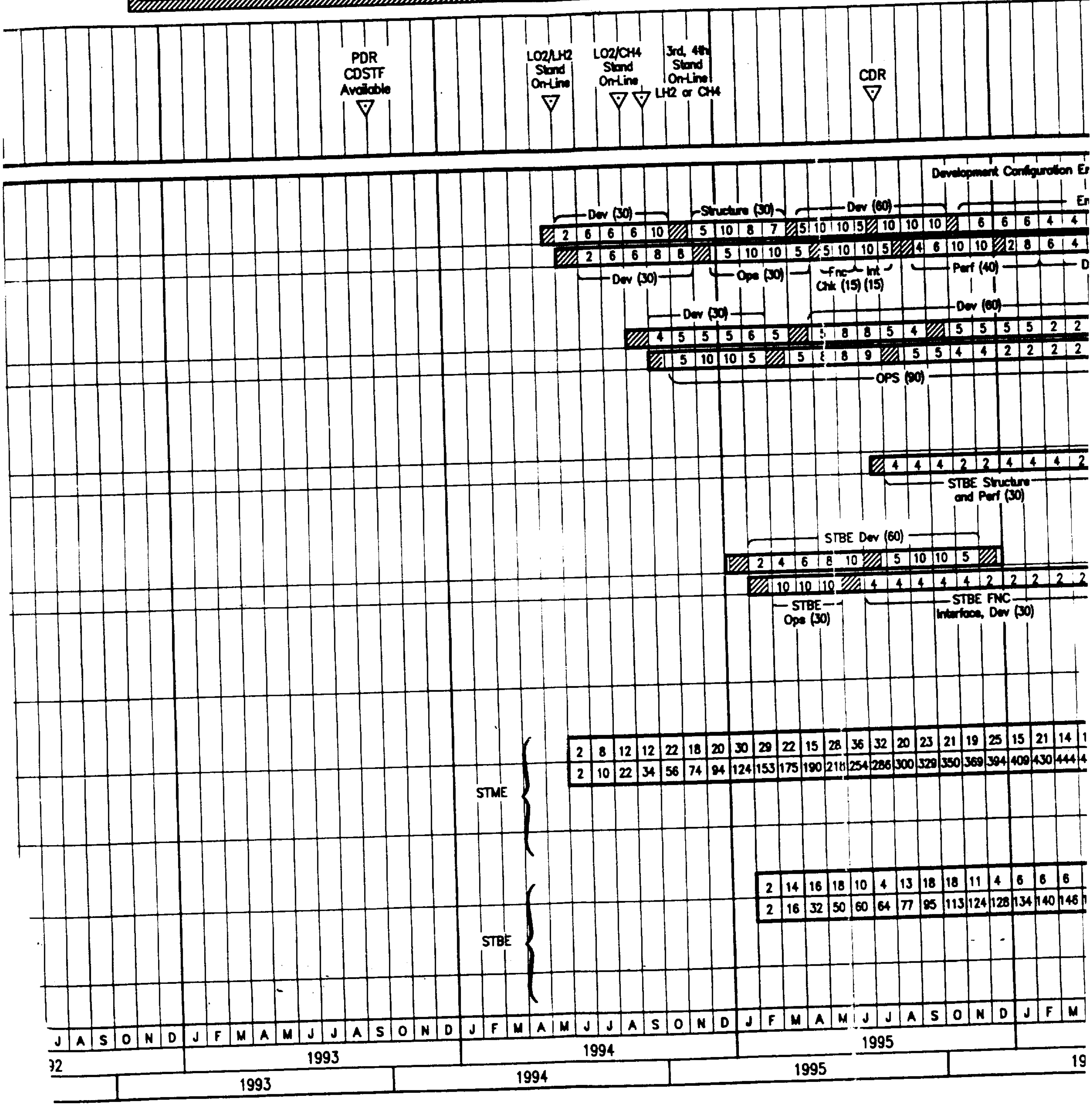
STME/DERIVATIVE STBE DEVELOPMENT SCHEDULE
90 MONTH/960 STME ENGINE FIRINGS/485 DERIVATIVE STBE FIRINGS



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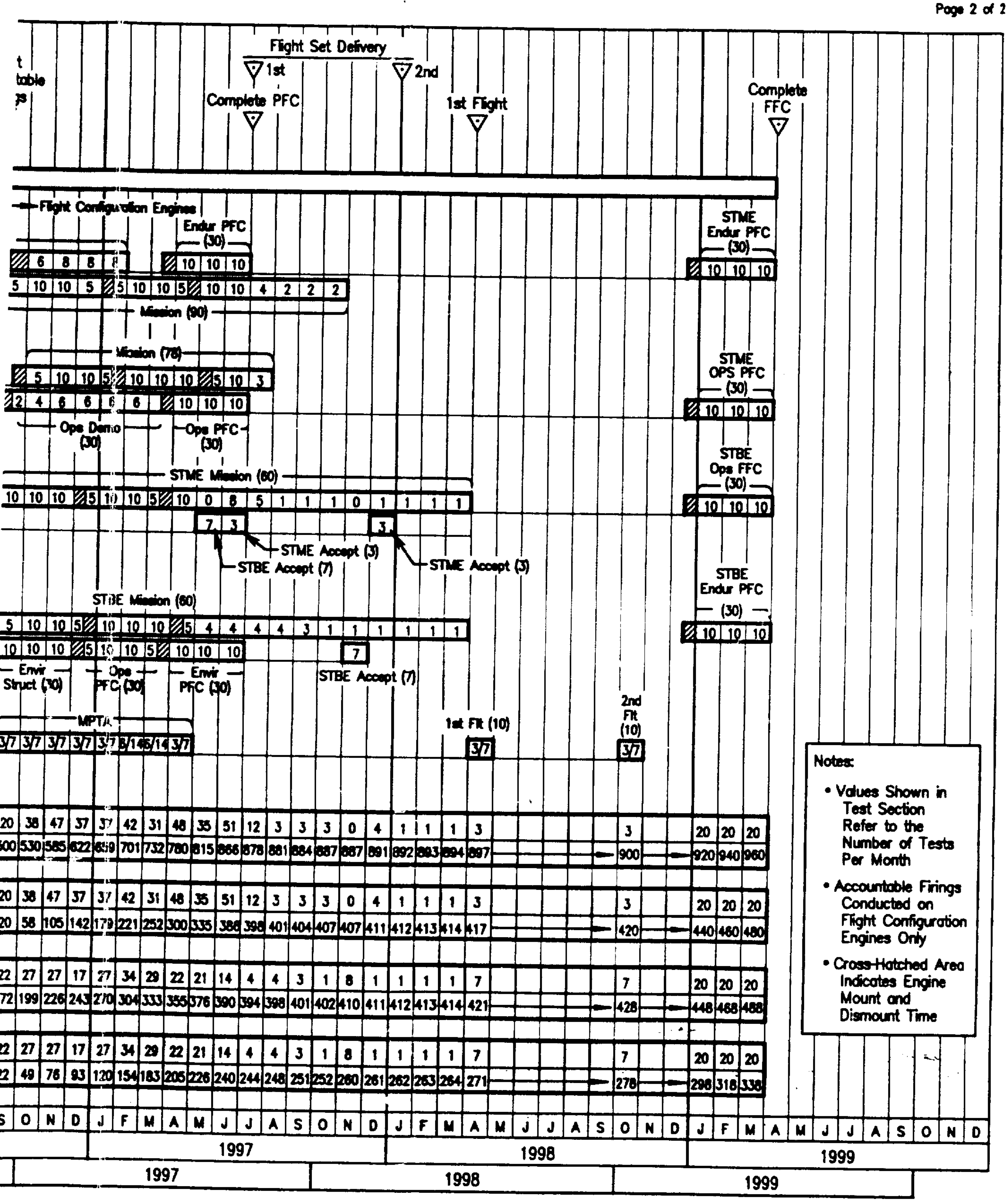

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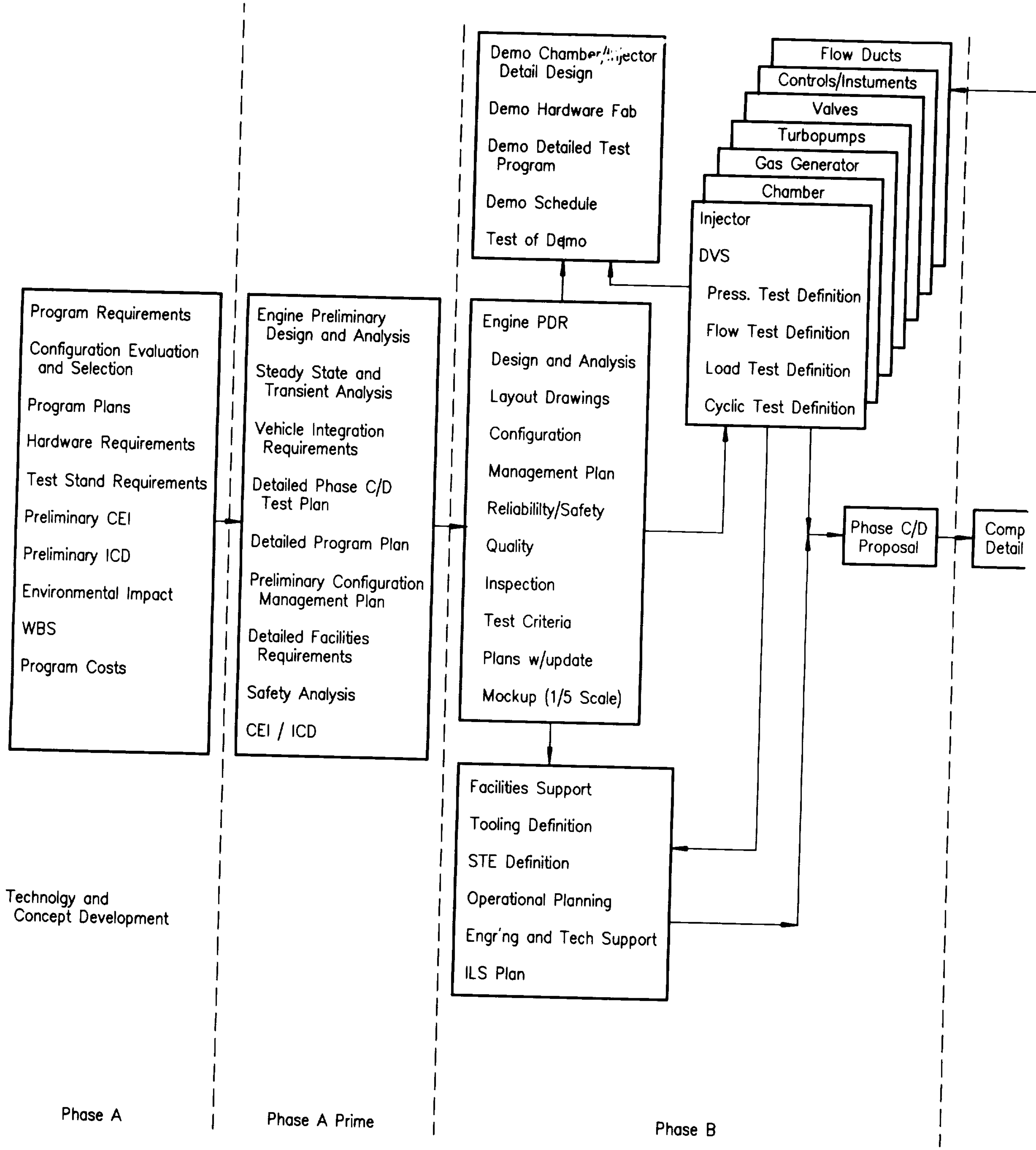

Figure 5.1-1. Logic Network

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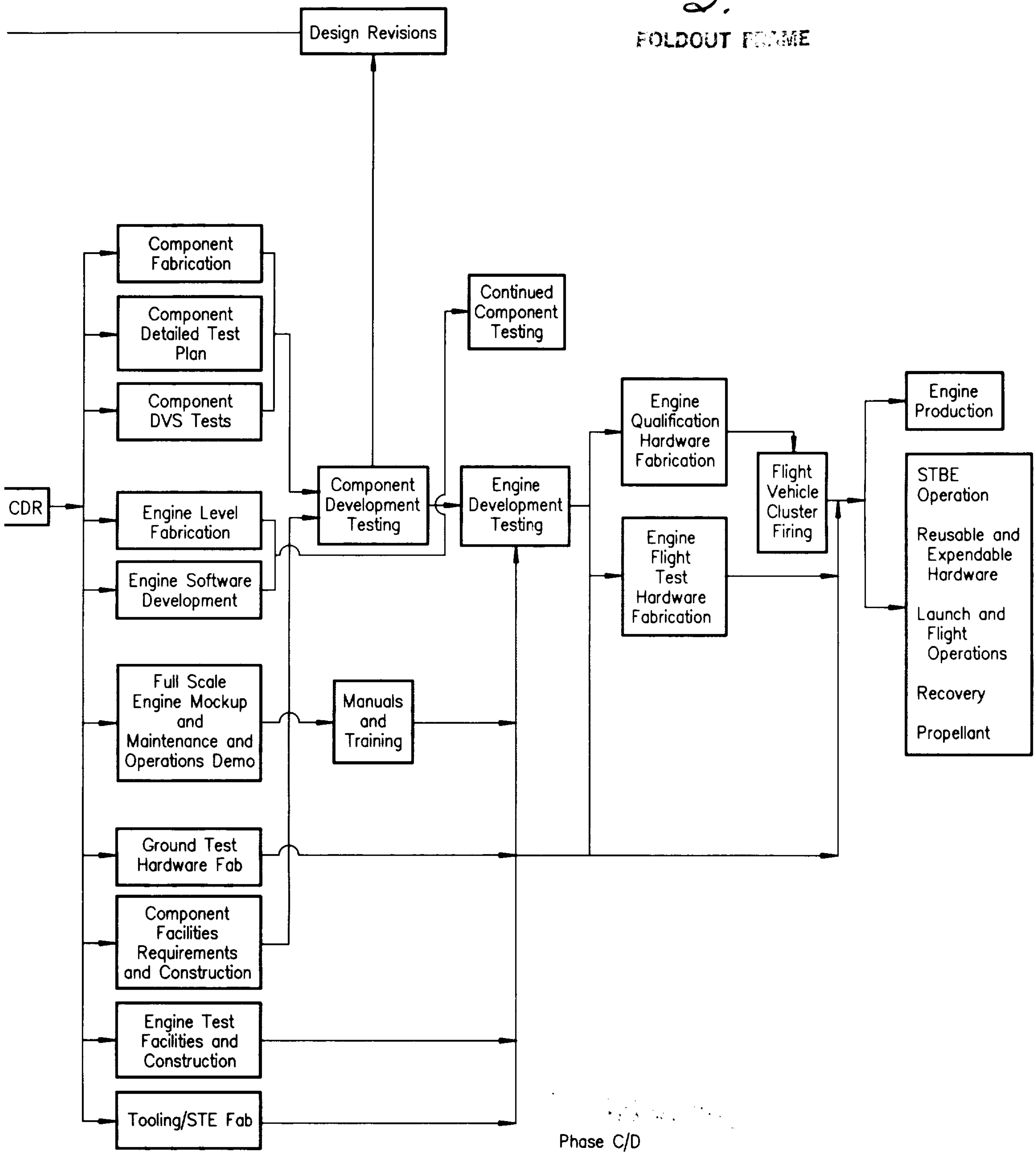

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As component fabrication is completed, component testing will be conducted. Information obtained during the component tests will allow design revisions necessary to optimize the hardware design. This provides a feedback loop into the DVS planning activity.

As a result and as a part of the detail design effort, the 1/5-scale mockup can be replaced with a full-scale mockup. This mockup greatly facilitates the design of the external flow ducting and allows a demonstration of engine maintenance and operation. The mockup and demonstrations done with the engine will allow the creation of manuals and training material.

The ground test hardware, tooling, and special test equipment necessary for engine testing and STBE operation will be fabricated during this phase.

Early in Phase C/D, the engine contractor must participate in the engine test facilities requirements and follow the test stand fabrication. When the initial tests of the component hardware are completed, the components can be assembled together to conduct engine development tests. Component tests will continue in parallel to accumulate confidence that test time-related malfunctions have been found and corrected.

As engine test time is accumulated and design iterations diminish, the design can be frozen and hardware for engine qualification and flight test can be fabricated.

One of the major elements during Phase C/D will be a firing of a cluster of engines with a stackup of vehicle tankage, etc.

The program then progresses into engine production and engine operation activities.

5.2 PROJECT SCHEDULES

5.2.1 Major Rig and Engine Tests

The Development Tests scheduled for the DDT&E Program are structured to evaluate and demonstrate all of the functional, durability and performance requirements of the engine. Initially, the component rigs (gas generator, main chamber assembly, LH_2 turbopump and LO_2 turbopump) lead the engine test to ensure that the component has sufficient performance, function and durability to qualify the component for integration into the engine. The rigs will also be used to evaluate part redesigns prior to introduction into a development engine. The rigs will be used in the development program up to the time that engine firings commence for the preliminary flight certification of the engine. At this time sufficient confidence should be demonstrated that the engine is safe to operate and any additional part changes can be evaluated in the engines. Table 5.2-1 shows the number of tests scheduled for the rigs for both the STME and derivative STBE engines. The number of component and engine tests for the derivative STBE are less than for the STME due to commonality of the majority of the hardware and also since the STME development program will lead the derivative STBE program. The commonality aspects of the derivative STBE are described in paragraph 4.1.1.

It should be noted that the number of rig tests on the derivative STBE LO_2 pump is limited by test facility capacity to 120 test runs. It is desirable to conduct 300 test runs of this pump since it has little commonality to the STME LO_2 pump and 300 runs are preferred when developing a new turbopump. In contrast, the derivative STBE fuel pump and gas generator are similar to the STME and they require fewer component tests than the similar STME component since the STME component will lead the development program.

Table 5.2-1. *Rig Development Tests*
(LO_2/LH_2 STME Engine)

Rig	Total Tests	Run Time Per Test	Total Test Time
TCA	150	7.5 sec	1,125 sec
GG	50	25	26,250
	100	250	
LO_2 Pump	70	25	59,250
	230	250	
LH_2 Pump	70	25	59,250
	230	250	
(LO_2/CH_4 Derivative STBE Engine)			
TCA	150	7.5 sec	1,125 sec
GG	60	250	15,000
LO_2 Pump*	20	25	25,500
	100	250	
LH_2 Pump	20	25	25,500
	100	250	

* Note: Tests limited by facilities capability.

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Several categories of test series are planned for the development of the STME and derivative STBE engine. The first engine test will follow the first rig test by eight months. The major test categories and test objectives are listed below.

Major Test Series	*Test Objectives*
• Functional Checkout	Leakage Tests Gimballing Capability Controller Checkout Health Monitor Checkout
• Interface	Gimbal Rate Tank Pressurization Propellant Inlet Purge
• Environmental/Structural	Acoustic Signature Engine Vibration Acoustic Loads Starting, Operating and Shutdown Loads Thermal Conditioning Component Stress and Vibration

• Operational Demonstration	Prestart Conditioning Ignition Start/Shutdown Rates, Impulse Throttle Command Response Combustion Stability Engine Pressure, Temperature, Flow Rates Engine Redline Limits POGO
• Performance Demonstration	Engine Calibration Thrust Level Specific Impulse Mixture Ratio Tolerance Performance Repeatability
• Development Testing	General development tests conducted on pre-preliminary Flight Certification Configuration engines to verify engine designs and to eliminate potential engine anomalies
• Mission Testing	Tests conducted on Preliminary Flight Certification engine configurations to demonstrate the reliability requirements of the engines. Firings conducted on these engine are all considered to be accountable firings.
• MPTA (Cluster) Tests	Fire all 10 vehicle engines at one time Verify base heating
• Preliminary Flight Certification Testing (PFC)	Sixty firings conducted on two engines to demonstrate durability and operability requirements of the engine specificaton.
• Development Flight Tests	Experimental flight test and booster engine recovery
• Final Flight Certification Test (FFC)	Sixty firings conducted on two engines to demonstrate final production engine durability and operability requirements of the engine specification. These tests follow the development flight tests.

To demonstrate reliability of the flight configured engine, all engine tests which contribute to the reliability demonstration of the engine must be conducted on hardware which has the configuration of the preliminary flight certification engines. These firings are termed accountable firings since they contribute to the reliability demonstration of the flight configured engine. To demonstrate the required 0.99 reliability at 90 percent confidence a total of 230 engine firings must be successfully accomplished without failure or malfunction of the engine which would require a premature engine shutdown. Alternatively, one malfunction could occur with a total of 388 firings and still meet the reliability requirement. The STME DDT&E Program has been

structured to be able to absorb one unanticipated engine failure requiring engine shutdown during the accountable firing phase of the development program without causing a development schedule impact. The derivative STBE program uses 264 accountable firings to demonstrate reliability requirements. Table 5.2-2 lists the engine tests and identifies the number of firings for each type of test. The total number of STME tests is 960 as specified by NASA, of which 414 are accountable firings that occur prior to first flight. The derivative STBE engine uses 488 total engine firings of which 264 firings are accountable prior to one first flight.

Table 5.2-2. *STME/Derivative STBE Development Tests*

	Total Firings		*Accountable Firings Prior to First Flight*	
Engine Tests	*STME*	*DERIV STBE*	*STME*	*DERIV STBE*
• Functional Checkout	15	10		
• Interface	15	10		
• Environmental/Structural	90	45	30	30
• Operational Demonstration	150	30	30	
• General Development (Pre-PFC Configuration)	230	70		
• Mission Testing (PFC Configuration)	258	90	258	90
• Performance Demonstration	40	15		
• Preliminary Flight Certification (PFC)	60	60	60	60
• MPTA	30	70	30	70
• Flight Test (With Checkout)	12	28	6	14
• Final Flight Certification (FFC)	60	60		
Subtotal	960	488	414	264
Total	1448			

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5.2.2 Development Schedules

A Summary Development Schedule of the STME/Derivative STBE Gas Generator engine and the detailed Development Schedule for this engine are shown in Figure 5.2-1. These schedules show the Advanced Development Program which precedes the start of full-scale development. Major milestones are listed at the top of the charts and the upper half of the development schedule shows the major component rig (GG, TCA, Turbopump) tests. The lower half of the chart shows the major engine development tests and the qualification tests.

Four engine test stands (each with two positions) are used for the STME/Derivative STBE Development Program. The Component and Subsystem Development Test Facility (CSDTF) is used for the Component Development Tests. One CSDTF test position is used for the GG, one for the TCA, two positions for the LH_2 turbopump, and two positions for the LO_2 pump. The maximum test rate was assumed to be eight firings (runs) per month for each position in the CSDTF and 10 engine firings per month for each engine test position. The typical firing rate is generally less than the maximum rate as shown in Figure 5.2-1. Table 5.2-3 shows a comparison of the maximum firing rate and the average firing rate for the STME/Derivative STBE Program.

Table 5.2-4 shows the number of tests scheduled for the engine for both the STME and Derivative STBE engines.

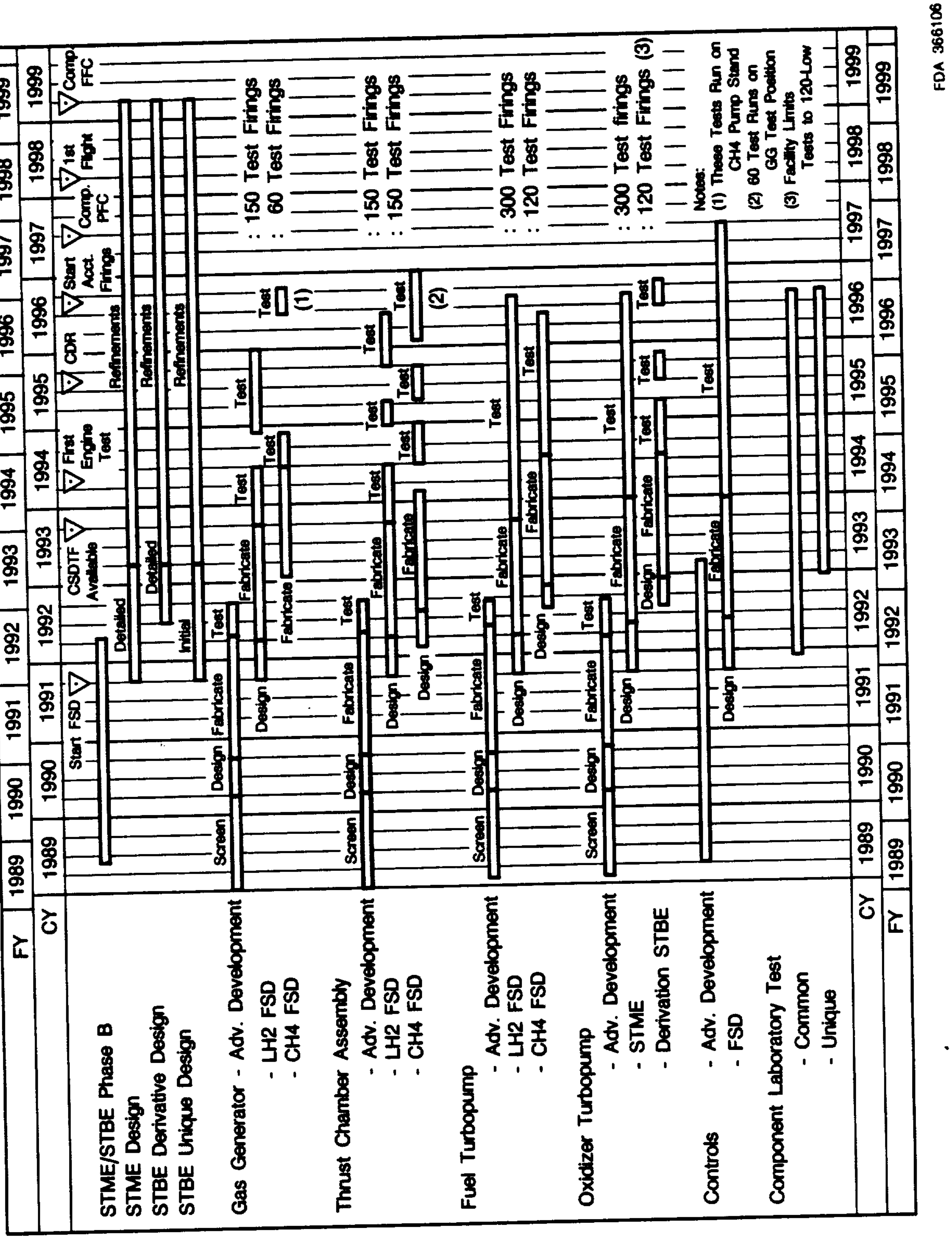

Figure 5.2-1. STME/Derivative STBE Development Schedule (Sheet 1 of 4)

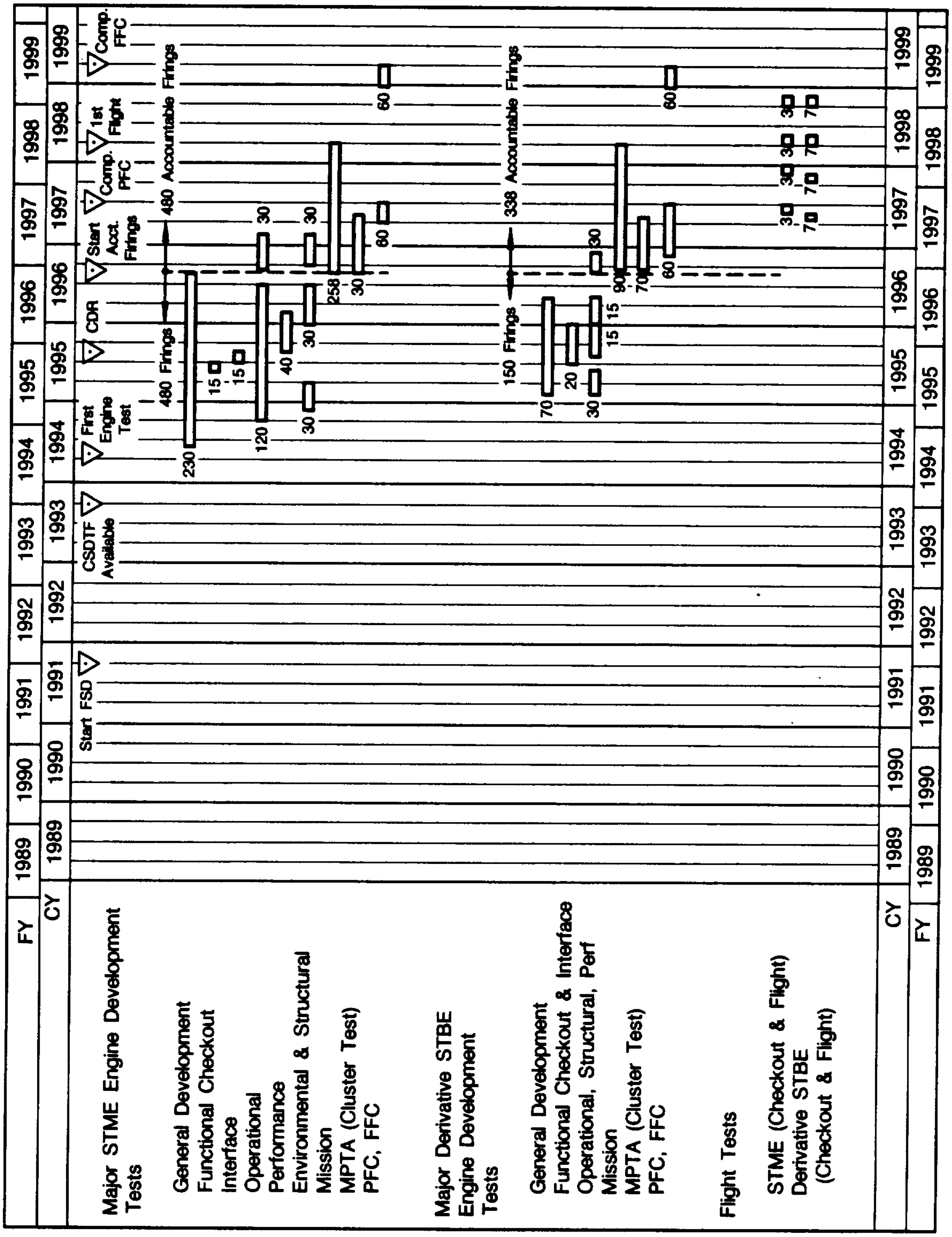

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Figure 5.2-1. STME/Derivative STBE Development Schedule (Sheet 2 of 4)

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Table 5.2-3. *STME/Derivative STBE Test Facility Usage Rates (Tests per Position)*

	Max Test Rate (Per Mo.)	Avg. Test Rate (Per Mo.) STME	Deriv STBE	Integrated
• CDSTF				
Thrust Chamber Assy	8	7.4	7.5	7.5
Gas Generator	8	7.8	7.5	7.7
LO_2 Pump (2 Positions)	8	7.6	7.9	7.7
LH_2 Pump (2 Positions)*	8	5.6	—	5.6
CH_4 Pump (1 Position)	8	—	5.4	5.6
• Engine Test (8 Positions)	10	6.4	6.1	6.3
• MPTA	2 Cluster Firing	1.1	1.1	1.1

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Table 5.2-4. *Engine Development Tests (STME and Derivative STBE Engine)*

Test	Total Firings	Run Time Per Test (sec)	Total Test Time (sec)
• Engine Development Tests			
STME	60	20	1,200
	140	360	50,400
	280	600	168,000
Derivative STBE	15	20	300
	30	160	4,800
	105	380	39,900
• Accountable Firing Tests Including MPTA and Certification			
STME	468	600	280,800
Derivative STBE	310	380	117,800
• Flight and Checkout			
STME (6 engines)	12	360	4,320
STBE (14 engines)	28	160	4,480
Total Test Time			
STME	960		504,720
Derivative STBE	488		167,280

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5.2.3 Hardware Requirements

The development tests are conducted with major component rigs and development engines which have a design life (with safety factors) of 120 firings. P&W intends to use each rig or engine for a total of 30 planned tests prior to removing the rig or engine from the test stand for overhaul and one reuse of an additional 30 firings. The rig or engine hardware will be returned from testing after a total of 60 firings to avoid the possibility of failure with hardware well past its operating life. Laboratory tests can be conducted with this hardware.

Table 5.2-5 shows the hardware requirements for the STME/Derivative STBE engine development programs.

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Table 5.2-5A. *STME Development Hardware Requirements (Gas Generator Engine Development)*

Rigs	*GG*	*TCA*	*LO_2 Pump*	*LH_2 Pump*
Number of Tests	150	150	300	300
New Rigs	3	3	5	5
Spare Rigs	1	1	2	2
Total New	4	4	7	7
Total Number of Equivalent Rigs for Rebuild (50% Replacement)	2	2	2.5	2.5
Total Number of Equivalent Rigs	6	6	9.5	9.5

Engines	*Development*	*Certification Tests*	*MPTA*	*Flight*
Number of Tests	798	120	30	12
New Engines	16	4	3	6
Spare Engines	2	0	1	2
Total New Engines	18	4	4	8
Total Number of Equivalent Engines for Rebuild (50% Replacement)	10.5	0	0	0
Total Number of Equivalent Engines	28.5	4	4	8

Total = 44.5 Equivalent Engines

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Table 5.2-5B. *Derivative STBE Development Hardware Requirements (Gas Generator Development Program)*

Rigs	*GG*	*TCA*	*LO_2 Pump*	*LH_2 Pump*
Number of Tests	60	150	120	120
New Rigs	1	3	2	2
Spare Rigs	1	1	1	1
Total New	2	4	3	3
Total Number of Equivalent Rigs for Rebuild (50% Replacement)	0	2	1.5	1.5
Total Number of Equivalent Rigs	2	6	4.5	4.5

Engines	*Development*	*Certification Tests*	*MPTA*	*Flight*
Number of Tests	270	120	70	28
New Engines	6	4	7	14
Spare Engines	2	0	2	5
Total New Engines	8	4	9	19
Total Number of Equivalent Engines for Rebuild (50% Replacement)	4	0	0	0
Total Number of Equivalent Engines	12	4	9	19

Total = 44 Equivalent Engines

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5.2.4 Summary

The proposed development schedule of 90 months for the STME/Derivative STBE GG engine is achievable with the NASA specified test facilities and availability dates. P&W plans to conduct 960 STME engine firings and 488 Derivative STBE engine firings to develop these engines. The CSDTF test capability for LO_2 turbopump testing may be marginal based on the assumed P&W test rates. All major program milestones as specified by NASA can be met for the development program.

5.3 FACILITY REQUIREMENTS

In the first Interim Report FR-19691-1, Volume II, Section 4.3, the results of the test stand tankage sizing study were presented for the tripropellant engine. Since that time, not only has the engine changed, but test facility contractors have now been put under contract for the three ALS test sites. This effort is being guided by the ALS Propulsion Test Facilities Working Group at NASA SSC. These contractors are designing and sizing the facilities to meet the requirements of the engine contractors.

Pratt & Whitney transmitted its engine requirements to the ALS Propulsion Test Facilities Working Group through a preliminary ALS Advanced Development Program, Liquid Propulsion, Component Interface Control Document (CICD). Table 5.3-1 lists the ALS Advanced Development Program components that will be tested at the government facilities. The two most important sections of the CICD addressed the interface points and the engine fluid requirements at the test article. Figures 5.3-1 through 5.3-8 show the interfaces required for the various components. Figures 5.3-9 through 5.3-19 show the fluid requirements, pressure, temperature, and flow rates for the major components. Three levels are presented for each requirement:

1. The expected maximum
2. An intermediate level that would be tested
3. A minimum level for starting.

Also, a preliminary estimate of the instrumentation requirements is presented in Table 5.3-2.

Table 5.3-1. *Components To Be Tested at Government Facilities*

• Injector, Main Combustion Chamber, Regeneratively Cooled Nozzle
• Subscale Injector and Calorimeter Combustion Chamber
• Gas Generator
• Subscale Gas Generator
• Turbopumps

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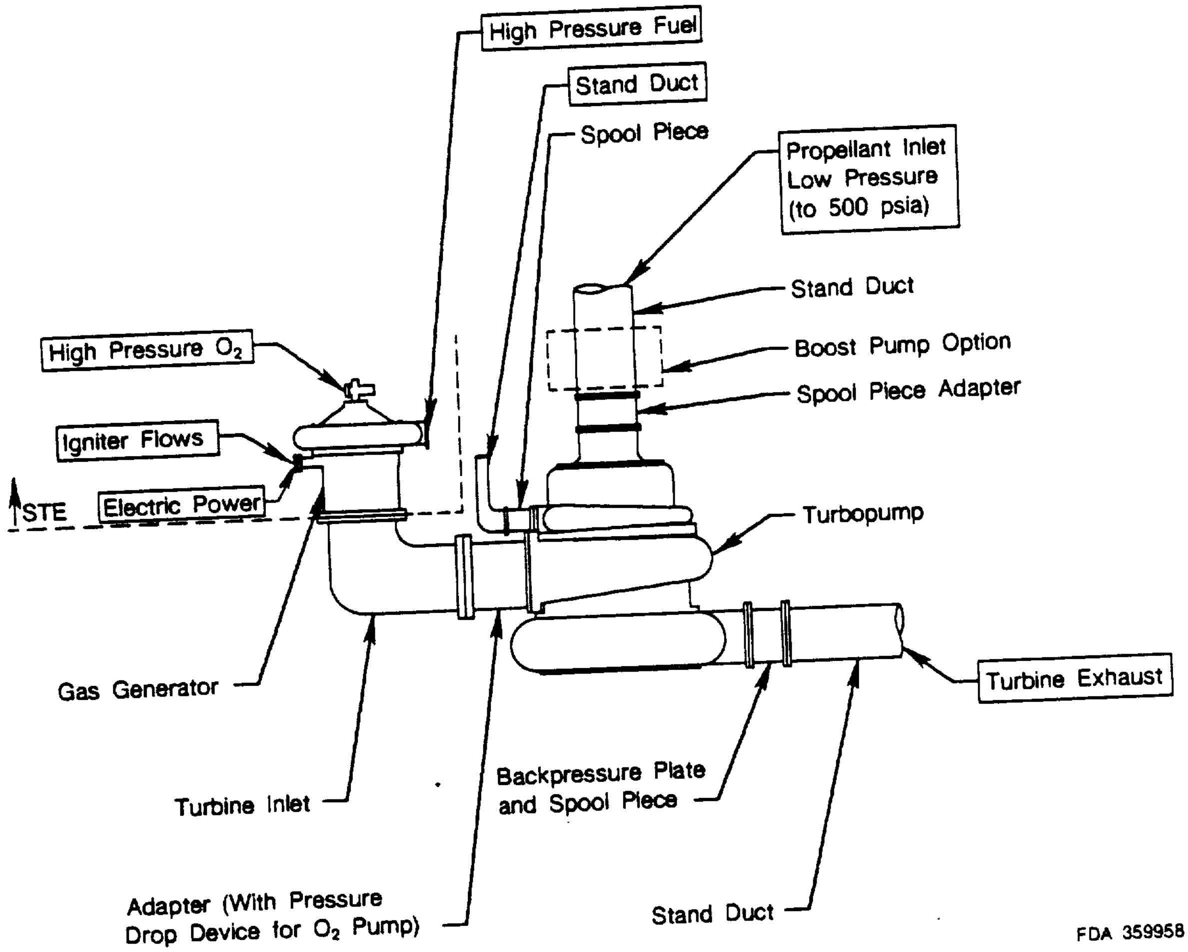

Figure 5.3-1. Turbopump Physical Interface Points

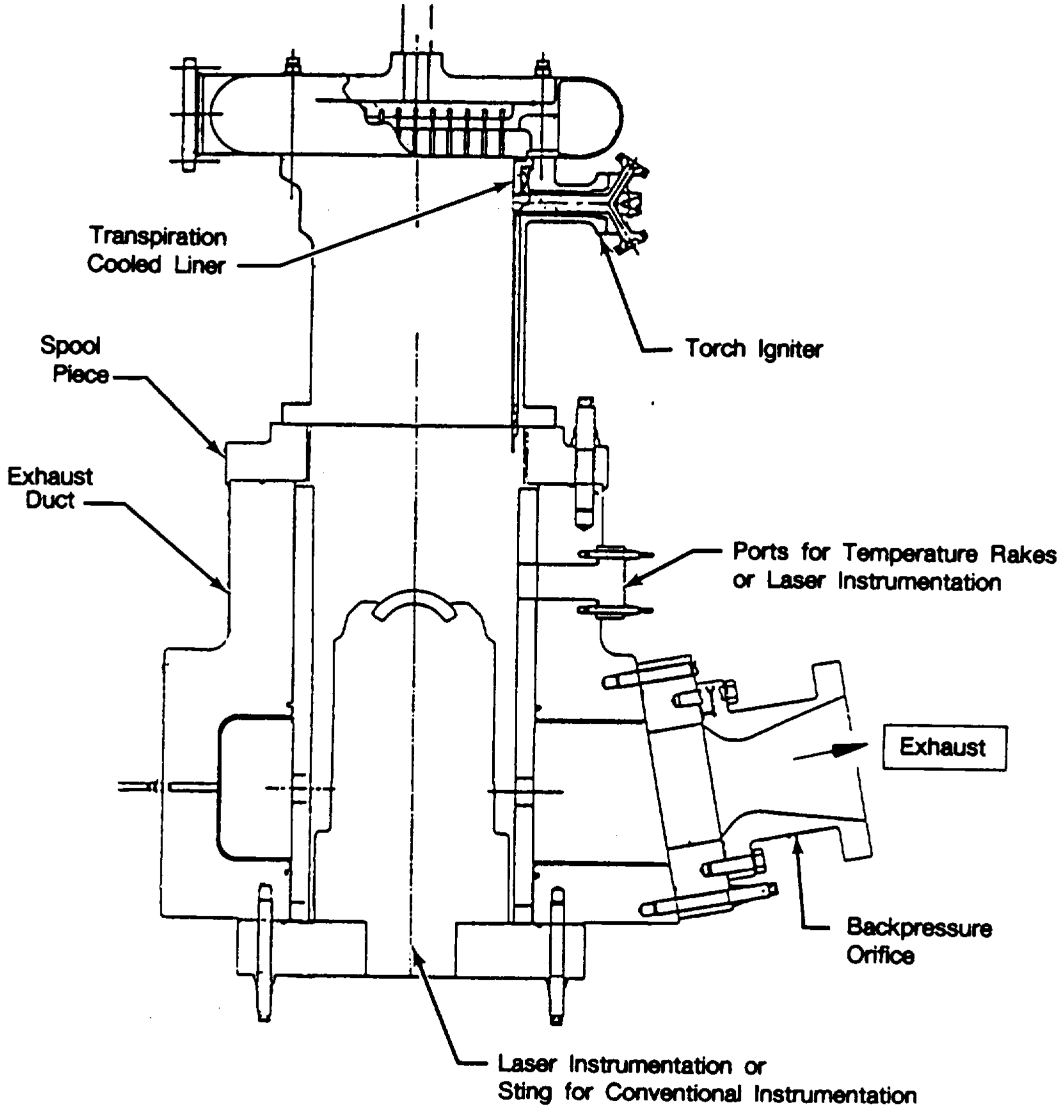

Figure 5.3-2. Gas Generator Physical Interface Points

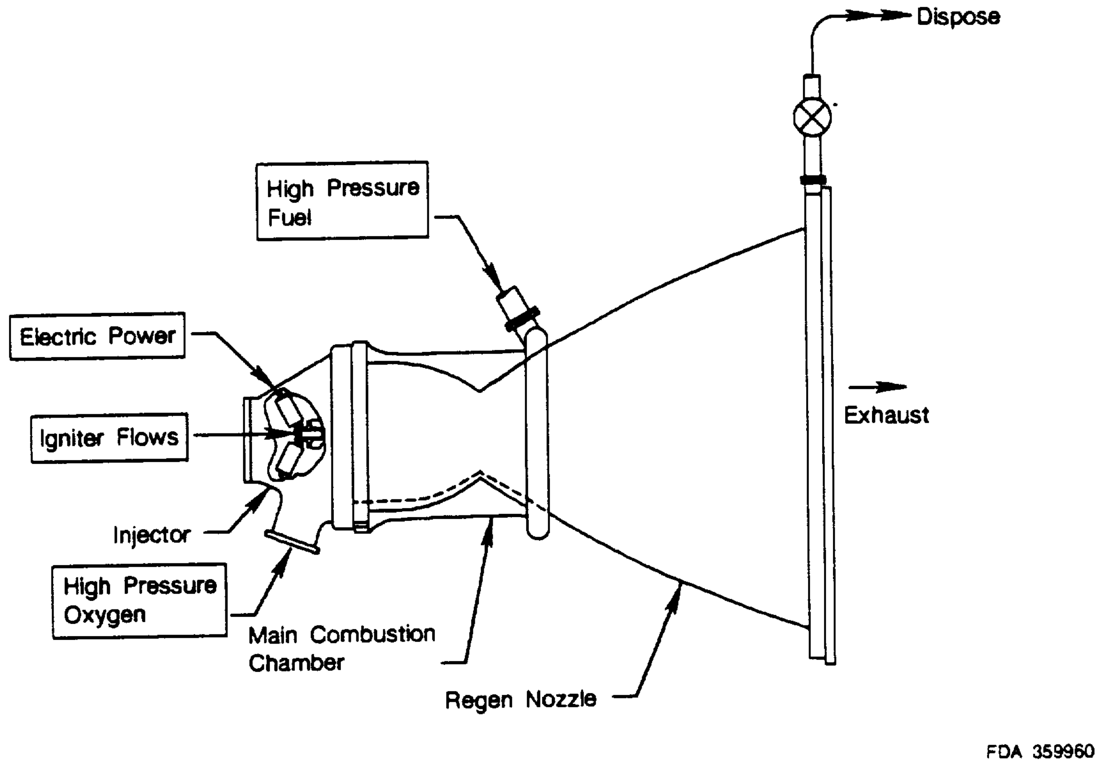

Figure 5.3-3. Regenerative Cooled Thrust Chamber Physical Interface Points

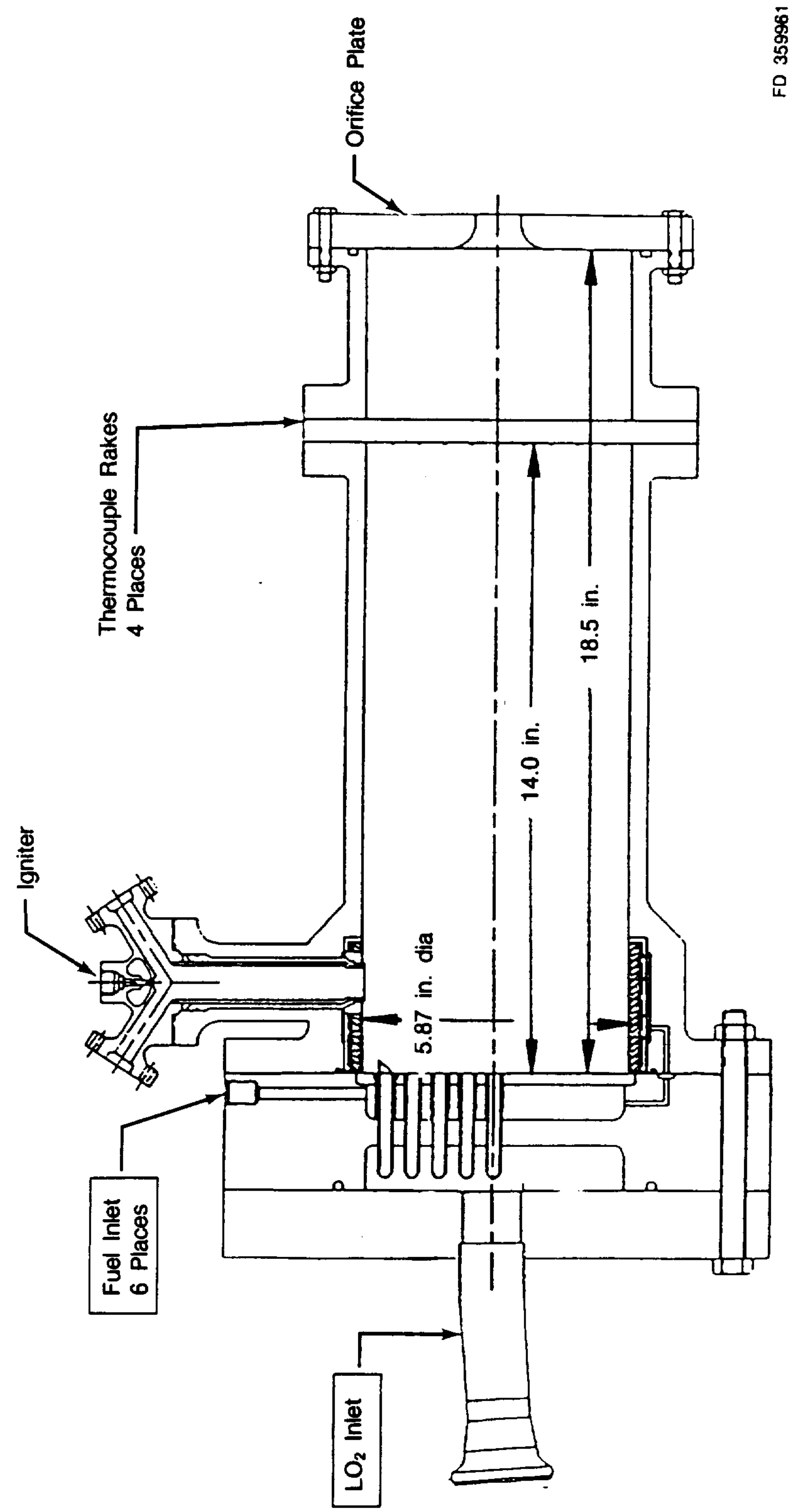

Figure 5.3-4. Subscale Gas Generator Physical Interface Points

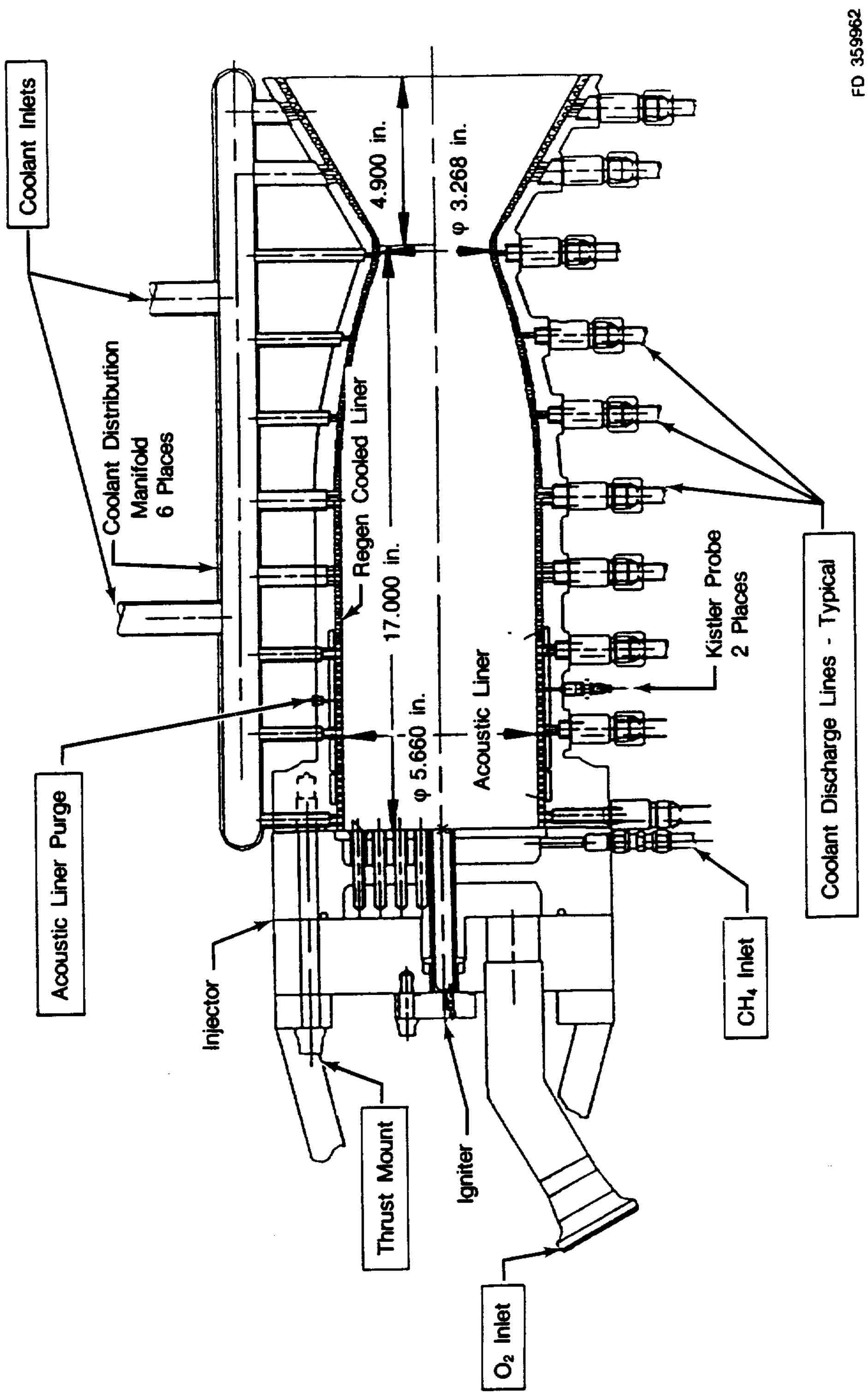

Figure 5.3-5. Subscale Thrust Chamber Physical Interface Points

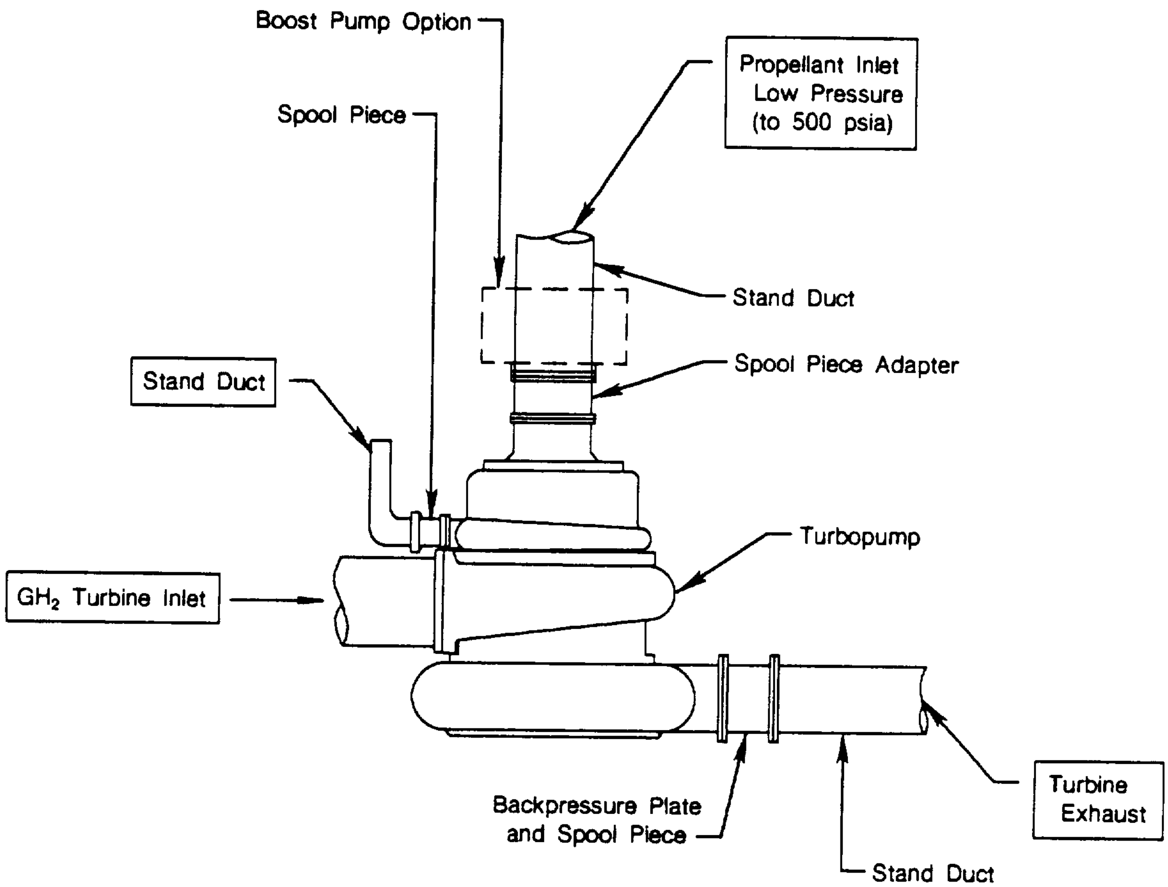

Figure 5.3-6. Split Expander LOX Turbopump Physical Interface Points

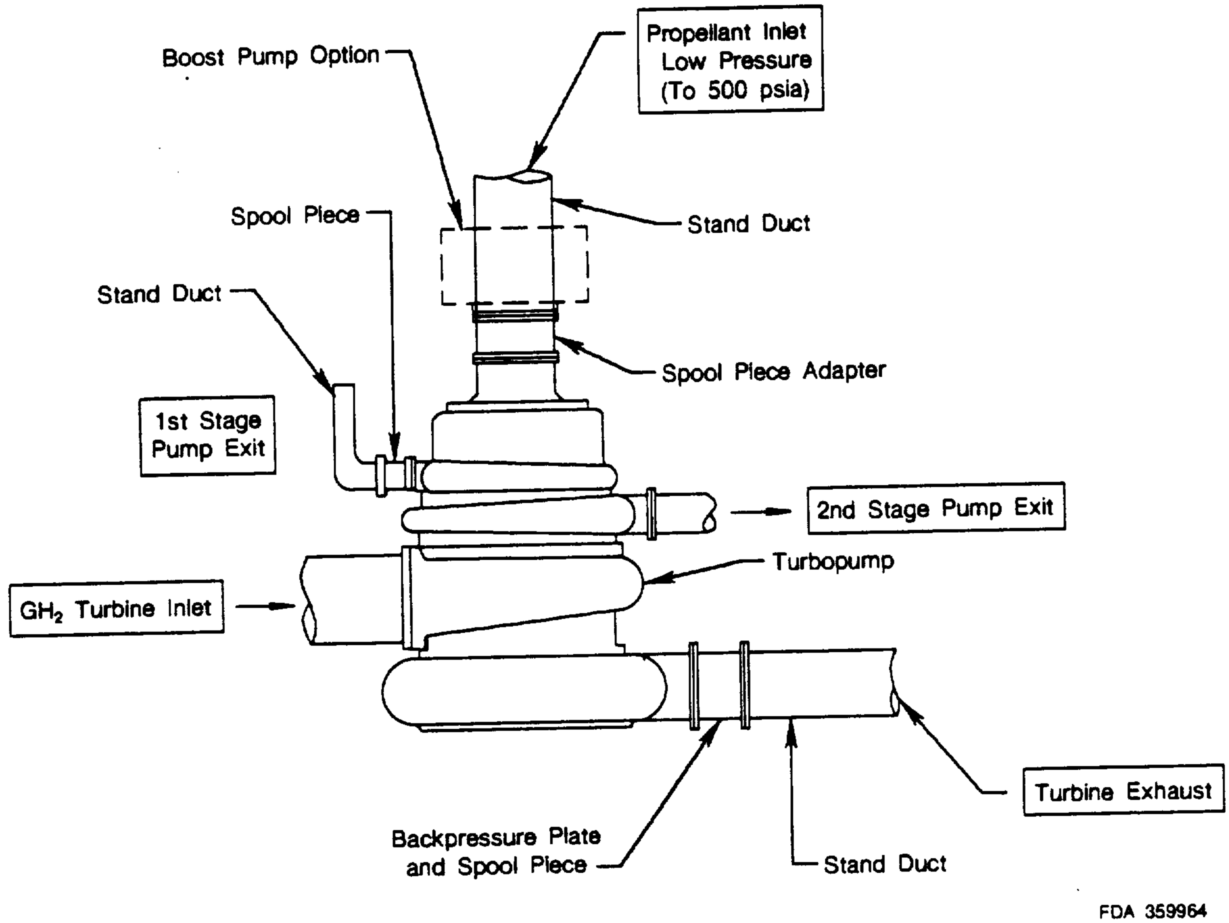

Figure 5.3-7. Split Expander Hydrogen Turbopump Physical Interface Points

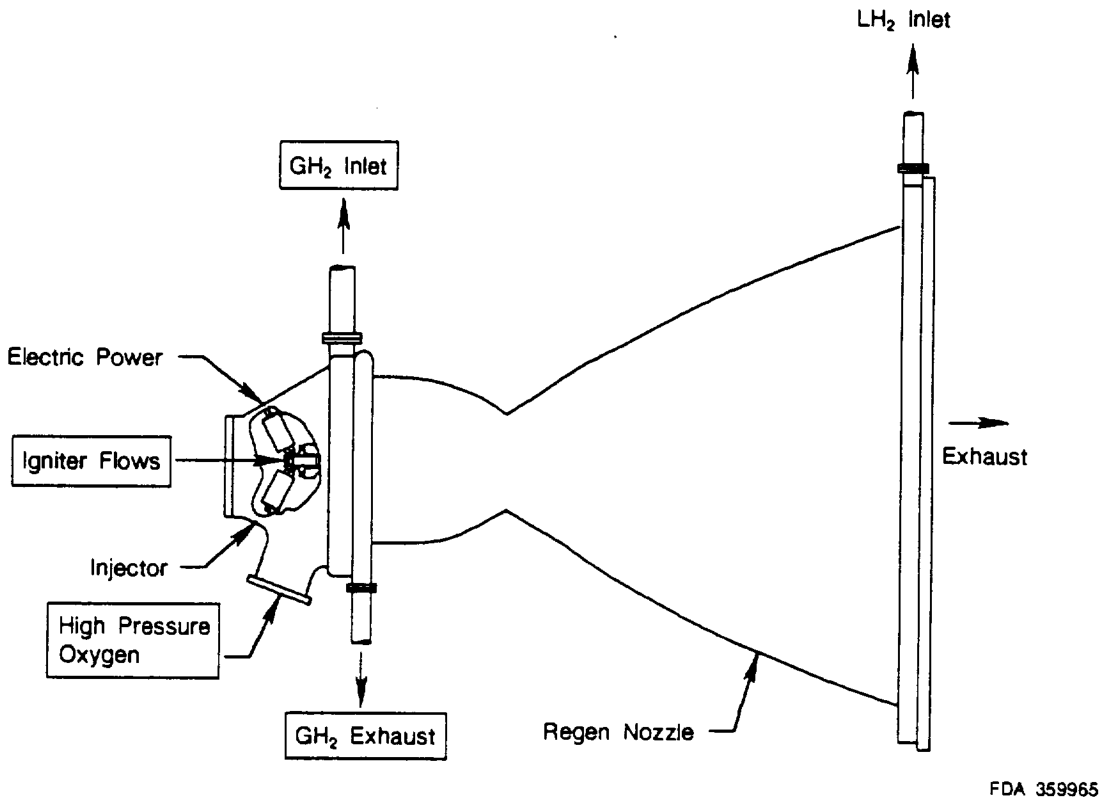

Figure 5.3-8. Split Expander Regenerative Cooled Thrust Chamber Physical Interface Points

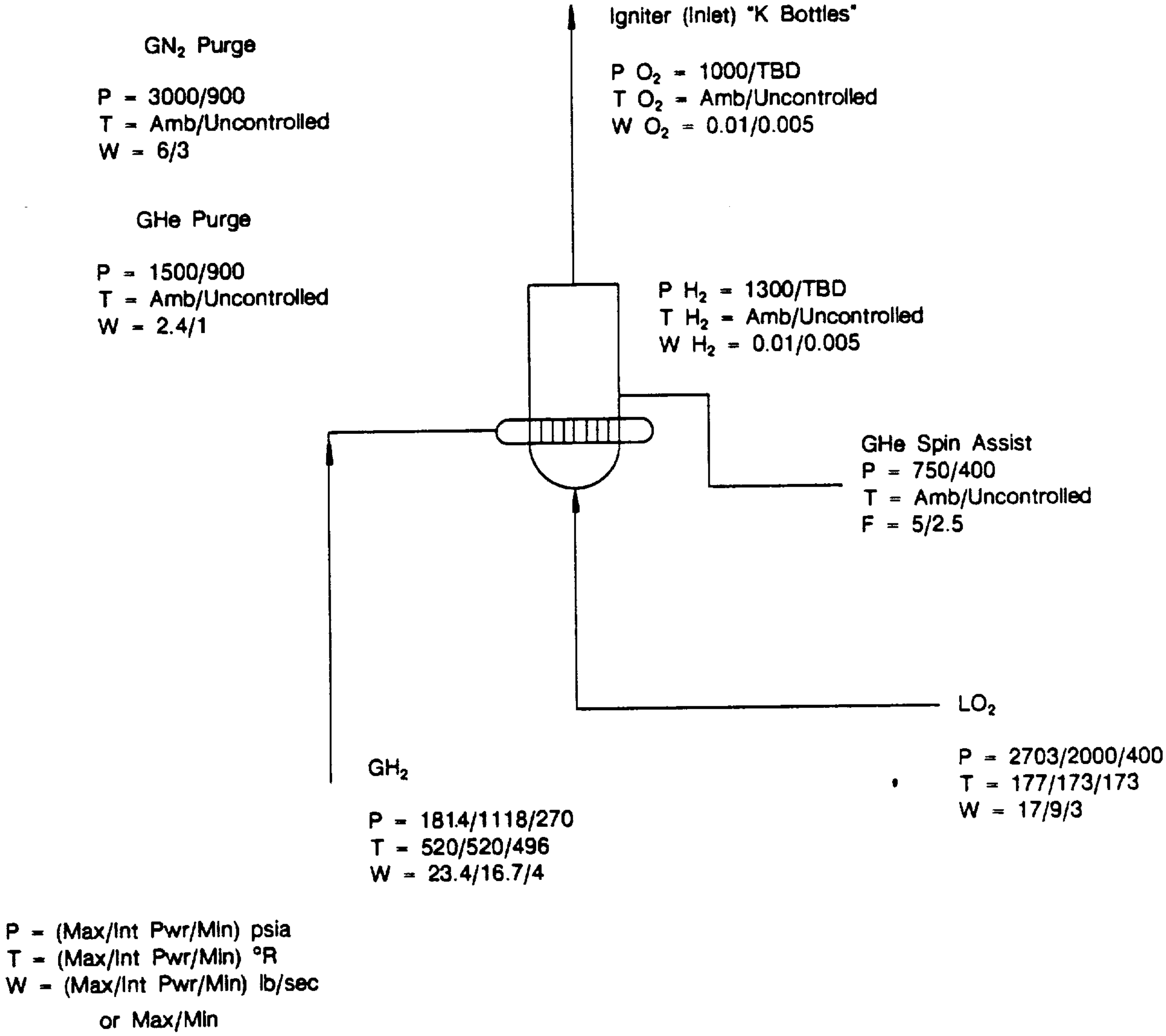

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Figure 5.3-9. Advanced Development Program Facility Interface Requirements for STME Hydrogen Gas Generator